DOT/FAA/AR-95/12

Office of Aviation Research
Washington, D.C. 20591

Effects of Stiffener/Rib Separation on Damage Growth and Residual Strength

May 1996

Final Report

U.S. Department of Transportation
Federal Aviation Administration

NOTICE

Technical Report Documentation Page

1. Report No.	2. Government Accession No.	3. Recipient's Catalog No.
DOT/FAA/AR-95/12		

4. Title and Subtitle

EFFECTS OF STIFFENER/RIB SEPARATION ON DAMAGE GROWTH AND RESIDUAL STRENGTH

5. Report Date
May 1996

6. Performing Organization Code

7. Author(s)
H.P. Kan and M. Mahler

8. Performing Organization Report No.

9. Performing Organization Name and Address
Military Aircraft Division
Northrop Grumman Corporation
One Northrop Avenue
Hawthorne, California 90250-3277

10. Work Unit No. (TRAIS)

11. Contract or Grant No.
Task 11, NAS1-19347

12. Sponsoring Agency Name and Address
U.S. Department of Transportation
Federal Aviation Administration
Office of Aviation Research
Washington, D.C. 20591

National Aeronautical and Space Administration
Langley Research Center
Hampton, Virginia 23681-0001

13. Type of Report and Period Covered
Final Report
September 1993 – April 1994

14. Sponsoring Agency Code
AAR-431

15. Supplementary Notes
Technical Monitor: P. Shyprykevich, William J. Hughes Technical Center
Administrative Support: M. Rouse, NASA LaRC

16. Abstract

Two existing composite aircraft structures were used to evaluate the effects of skin/stiffener separation on the residual strength of the structures. These structures are basically compression dominated upper wing structures designed to comply with the impact damage tolerance requirements. The severity of impact damage and delaminations were analytically compared with that of skin/stiffener disbond. Critical disbond sizes were determined so that the residual strengths of the structures are comparable to those obtained from impact damage tolerance designs. A damage tolerance certification approach based on the results of this study was recommended. The approach is to prevent local buckling in the disbond region under the applied load that governs the damage tolerance design for impact damage and delaminations. This would lead to a critical disbond length for the structure that has the same residual strength capability as in the case of impact damage and delamination.

17. Key Words
Composites, Damage Tolerance, Impact, Disbond, Delamination

18. Distribution Statement
This document is available to the public through the National Technical Information Service, Springfield, VA 22161

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of Pages	22. Price
UNCLASSIFIED	UNCLASSIFIED	61	

Form DOT F 1700.7 (8-72) Reproduction of completed page authorized

PREFACE

This report was prepared by the Northrop Grumman Corporation, Military Aircraft Divisions, Hawthorne, California, covering work performed under Task 11 of NASA Contract No. NAS1-19347 between September 1993 and April 1994. This specific task was conducted under an Interagency Agreement between the William J. Hughes Technical Center, Atlantic City International Airport, New Jersey and the National Aeronautical and Space Administration Langley Research Center, Hampton, Virginia. Technical direction was provided by P. Shyprykevich, William J. Hughes Technical Center, with the advice of J. Soderquist, FAA Headquarters. Administrative support was provided by M. Rouse, NASA Langley Research Center.

The work was performed in Northrop Grumman's Structural Integrity and Materials Technology Department under the overall supervision of Dr. R.B. Deo. Dr. H.P. Kan was the Principal Investigator with analysis support of M. Mahler and documentation support of R. Cordero.

TABLE OF CONTENTS

LIST OF ILLUSTRATIONS

LIST OF TABLES

EXECUTIVE SUMMARY

Compliance with Federal Aviation Regulations FAR 25.571 for fail safety in commercial aircraft is typically demonstrated by analysis and test of stiffened metal panels with a two-bay crack extending through the stiffener separating the two bays. A similar compliance methodology for composite aircraft structures does not exist at the present time. The objective of this task was to define the type, extent, and location of damage which would meet the requirements for composite structures in commercial aircraft similar to those in FAR 25.571 for metal structures.

Two existing composite aircraft structures were selected to demonstrate the certification methodology developed in this study. Both structures were designed to satisfy the impact damage tolerance requirements. These structures are basically soft wing skins with bonded stringers. This design feature makes skin/stiffener separation a potential damage mode that threatens the integrity of the structure. A transport wing was selected as a representative structure for a large airplane, and a military aircraft wing was selected as a representative structure for small airplane.

Three competing damage types were considered in this study. They were: impact damage, delaminations, and skin/stiffener disbonds. The influence of this damage types on the residual strength of the two existing composite structures were analytically determined. The severity of impact damage and delaminations were analytically compared with that of skin/stiffener disbonds. Critical disbond sizes were determined so that the residual strength of the structures were comparable to those obtained from impact damage tolerance designs.

The results of this study indicated that, for typically designed composite wing structures, a completely disbonded stringer represented the most severe damage scenario among the damage types considered. This type of damage mainly affects bonded or cocured structures under predominantly compression loads. The local strength at the damaged location, depends on the design details of the structure, may be significantly lower than the residual strength due to impact damage. Because of the large strength reduction, damage tolerance design based on such a damage scenario would impose a significant weight penalty to the structure. In order to achieve an efficient structure design without sacrifice the structural integrity a partial skin/stringer disbond is recommended as a damage tolerance certification requirement.

A damage tolerance certification approach based on the results of this study was recommended. The approach is to prevent local buckling in the disbond region under the applied

load that governs the damage tolerance design for impact damage and delaminations. This would lead to a critical disbond length for the structure that has the same residual strength capability as in the case of impact damage and delamination.

SECTION 1

INTRODUCTION

Federal Aviation Administration (FAA) requirements for damage tolerance and durability of commercial aircraft structures are contained in Federal Aviation Regulations 25.571, reference 1. In general, it requires that catastrophic structural failure due to fatigue, corrosion, or accidental damage must be avoided throughout the operational life of the airplane. Guidelines for an acceptable means of complying with this regulation are given in the FAA Advisory Circular AC 25.571-1A, reference 2, for metal structures.

Compliance with FAR 25.571 for fail safety in commercial transport aircraft is typically demonstrated by analysis and test of stiffened metal panels with a two-bay crack extending through the stiffener separating the two bays. This compliance methodology evolves from extensive experience and data for tension dominated metallic structures, and it provides a high level of confidence for structural integrity.

A similar compliance methodology for composite aircraft structures does not exist at the present time. An extensive database is being developed under the NASA Advanced Composites Technology (ACT) programs, references 3 through 8. Under these programs, large cracks and low-velocity impact damage are identified as potential threats to the structural integrity. The large-crack scenario has evolved from experience gained from metallic structures and it is intended to simulate accidental damage, such as uncontained engine blade impact. Due to the complex failure modes involved in damaged composite structures, the results obtained in these programs generally provide a database for specific structural design. In reference 9, damage tolerance requirements for composite military aircraft structures are addressed. Effects of impact damage on structural designs were extensively studied under this USAF/Boeing/Northrop program. A semiempirical strength prediction method was developed to assist impact tolerance design of composite structures.

In addition to large cracks, delaminations, and impact damage, skin/stiffener separation in bonded or cocured composite structures is a form of damage that needs to be addressed during the certification process of aircraft structures. This is discussed in reference 10. This type of damage may exist as manufacturing defects or be induced from repeated loading. Skin/stiffener disbond is most critical in compression dominated structural members because of the reduced structural stability. Furthermore, this type of damage normally escapes visual inspections due to its location. Therefore, a certification methodology must be established to assure the structural integrity when partial or complete skin/stiffener disbond occurs.

The objective of this task was to define the type, extent, and location of damage which would meet the requirements for composite structures in commercial aircraft similar to those in FAR 25.571 for metal structures.

Section 2 of this report describes the two existing composite structures used in the analytical studies of this program. These structures are basically compression-dominated upper wing structures designed to comply with the impact damage tolerance requirements. A Boeing transport aircraft wing, which was studied extensively under the USAF/Boeing/Northrop Damage Tolerance of Composites program (reference 9), is used as a representative structure for a large airplane. Another structure, typical of military aircraft wings designed for impact damage tolerance, was selected as a representative structure for a small, general aviation aircraft. During design process large-area skin/stiffener disbond was not considered as a design requirement for either of these structures. Because of the design feature of these structures, manufacturing or operationally induced disbonds between the skins and the stiffeners of these structures are highly likely. This makes these structures ideal examples for the current investigation.

In section 3, damage scenarios that threaten the integrity of the two structures are discussed. For this type of structure, low-velocity impact is considered as the baseline damage mode. Delamination of the skin laminate was also considered, even though this mode of damage is generally less severe. The severity of impact damage and delaminations are then compared with that of skin/stiffener disbond.

Section 4 outlines the analysis conducted during the performance of this task. Because large-area disbond between the skin and stiffener generally results in skin and stiffener acting independently, the major concern in the damaged structure is the loss of structural stability. Local and global buckling analyses were performed on the structures with and without disbond. Critical disbond sizes were determined so that the residual strengths of the structures are comparable to those obtained from impact damage tolerance designs. In addition, for comparison purposes, critical delamination sizes were determined, using the method developed in reference 9.

A damage tolerance certification approach based on the results of this study is outlined in section 5. For the type of structure considered, three competing damage types govern the damage tolerance design. They are impact damage, delamination, and skin/stiffener disbond. In the cases of impact and delaminations, the damage tolerance design criterion is traditionally based on inspection capability, such as barely visible impact damage with an impact energy cut-off (100 ft-lb) and a 2-inch-diameter circular or equivalent area delamination. Certification approaches for impact damage and delaminations are discussed in references 11 and 12, respectively. In the case of skin/stiffener disbond, the damage is not visually detectable from the exterior of the structure. In addition, disbond growth is likely if local buckling has occurred. Therefore, a reasonable approach is to prevent local buckling in the disbond region under the applied load that governs the damage tolerance design for impact damage and delaminations. This would lead to a critical disbond length for the damaged structure that has the same residual strength capability as in the cases of impact damage and delaminations. Such an approach is discussed in section 5. Finally, the conclusions drawn and the recommendations made based on the results of this research are summarized in section 6.

SECTION 2
STRUCTURAL DESCRIPTIONS

Two existing composite aircraft structures were selected to demonstrate the certification methodology developed in this study. Both structures were designed to satisfy the impact damage requirements. These structures are basically soft wing skins with bonded stringers. This design feature makes skin/stiffener separation a potential damage mode that threatens the integrity of the structures. A Boeing transport wing, studied under the Damage Tolerance of Composites program (reference 9), was selected as a representative structure for a large airplane, and a V-22 wing was selected as a representative structure for a small airplane. The arrangements of these structures are briefly described in the following paragraphs.

2.1 LARGE-AIRPLANE WING STRUCTURE

The baseline aircraft used in reference 9 is the Boeing C-X demonstration transport. This transport is a three-engine turbofan aircraft capable of airlifting a substantial payload over intercontinental ranges. It is designed to support maximum operational utility and reliability with minimum structural maintenance. Emphasis is on structural simplicity and ease of access for inspection and routine maintenance. Its size and wing loading are generic to a majority of large aircraft.

The C-X transport wing comprises three primary sections: a constant center section portion and left and right sections that taper in both planform and thickness. The section splice occurs outboard of the wing-mounted engine nacelle so that the center section incorporates the engine support structure, body attachment, and the upper surface blown-flap systems. The basic wing box is a two-spar configuration with multipanel upper and lower skins that are stiffened by stringers and ribs. The average stringer spacing is 5.80 inches on the upper panels and 6.76 inches on the lower panels. Rib spacing is 29.0 inches in the center section and 28.0 inches in the outboard sections.

The full-scale test wing box used in reference 9 was designed to account for all loading conditions pertinent to design of the actual wing structure. Primary design emphasis was on the upper surface panel as this component had the greatest weight impact on impact damage tolerance.

The primary composite components of the design are the upper and lower surface panels, channel section front and rear spars, and two ribs: one intermediate rib and the other a shear-tied rib. Other smaller parts consisted of stiffeners on the ribs and spars and shear clips to transfer the load between the various box elements. The load introduction fittings and the two end closure ribs are metal.

The wing box surface panels are designed to an end load of 25 kips/inch. The ultimate design strains for the box, excluding environmental effect factors, are 0.006 in/in for tension and compression and 0.012 in/in for shear for the undamaged skins. The maximum strain for damaged skin is 0.0032 in/in, and the residual strength requirement is 0.004 in/in. The box is fabricated from the Hercules AS4/3501-6 graphite-epoxy material system.

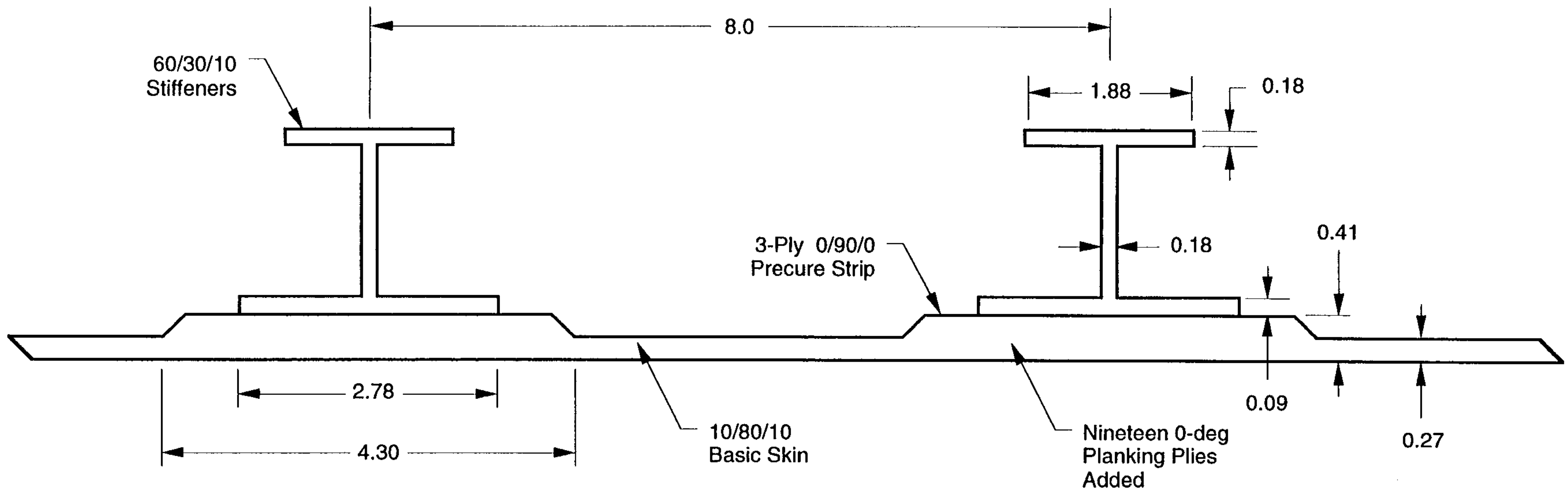
The surface panels featured relatively soft 10% of 0°, 80% of ±45°, and 10% of 90° plies, (10/80/10) skins, with additional 0-degree plies, identified as planking, interleaved in the basic skin at each stiffener location. The panels are stiffened by discrete (60/30/10) I-section stiffeners. Because both the upper and lower surface panels are designed to the same strain, panel details are identical. The dimensions of the test box are 96 inches long, 48 inches wide, and 29 inches deep.

A schematic of the details of the upper and lower surface panel basic section is shown in figure 1. The basic (10/80/10) skin is soft because of the predominance of ±45 degree plies. Axial load-carrying reinforcement is concentrated in unidirectional strips interleaved in the skin under each stiffener and in the stiffeners themselves. The soft skin is highly tolerant of damage. The planks at the stiffeners are damage resistant because of their increased thickness. Because stiffeners are internal, they receive little exposure to damage threat and are therefore not damage critical.

2.2 SMALL-AIRPLANE WING STRUCTURE

A typical wing of a military aircraft was selected as a representative structure for a small airplane. This composite wing is a single structural unit from tip to tip. The wing is composed of a main single-cell torque box, fixed trailing edge, wing/fuselage attachment fittings, flaperon, and leading edge. The wing is configured to support a pylon/nacelle assembly at each end and is attached to the fuselage through a fold stow mechanism.

The single-cell wing torque box assembly consists of upper and lower I-section stiffened skins, forward and aft spars, and eighteen ribs. All the components except the two tip ribs at each end are made of IM6/3501-6 carbon/epoxy tape material. The two tip ribs at each end are made of forged 7050 aluminum alloy. Small doors are provided in the lower skin to permit local access and fuel cell installation.



NOTES:

- Material: AS4/3501-6 Tape, 0.0074-in Ply
- Dimension in Inches

F94-HPK/03

FIGURE 1. UPPER AND LOWER SURFACE PANEL BASIC SECTION OF THE BOEING DAMAGE TOLERANT WING TEST BOX.

The basic skin of the wing box upper panel varies from 19-ply (5/84/11) layup to 21-ply (5/85/10) layup. The fibers are skewed at a 6-degree angle off the spanwise direction due to wing sweep. Bundles of 3.5-inch-wide, 0-degree plank plies are interleaved in the basic skin along each stringer and spar chord centerline to provide additional axial load-carrying and damage tolerance capability. A 3.5-inch-wide ply of adhesive is also laid up on each side of the plank ply bundles to improve damage tolerance. The basic skin near the wing tip is extensively padded up due to high local loads at the tip from the pylon. The basic skin is also padded around the wing/fuselage interface to reduce the strain level at the sweep and dihedral discontinuity. Five I-section stringers are fabricated and cobonded to the skin. Typical skin/stringer cross section is shown in figure 2. A ply of precured fabric is placed under the attached flange of each stringer to protect the skin from damage when removing/replacing stringers. Near the wing/fuselage interface, the stringers and planks are lap-spliced to accommodate the sweep and dihedral angles.

The lower surface panel is similar to the upper surface panel in concept and configuration, but differs in several significant details. Unlike the upper surface panel, the lower surface panel does not have adhesive added to the outside of the plank bundles. The lower surface panel contains large and small access holes. The access hole regions are padded up to account for local stress concentrations.

The design criteria for the composite components of the wing structure are summarized below:

1. Static strength requirements:
   No failure at ultimate loads
   Linear to failure
2. Skin, spar, and rib webs may buckle beyond limit load
3. Stiffeners, stringers, and caps are unbuckled to ultimate load
4. Clearly visible impact damage
5. Environment:
   Temperature: -65°F to 160°F
   Humidity, salt spray, snow, rain, hail, sand/dust
6. Fail-safe: redundant load paths where possible
7. Ballistic: limit load strength and safe continuance of flight for 5 hours and safe landing

8. Fatigue:

   Design analysis of four lifetimes

   Fatigue test of two lifetimes

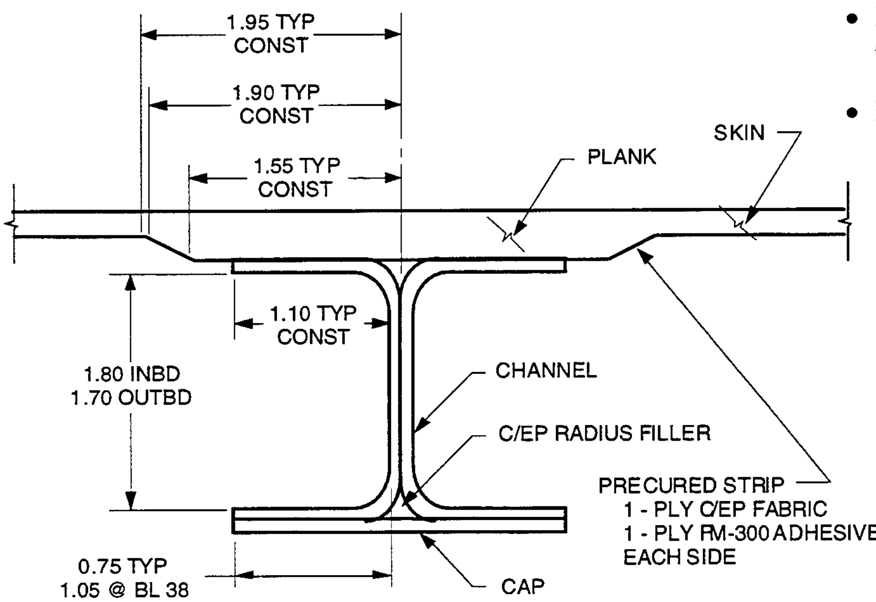
9. Damage tolerance:

   Maximum NDT accepted damage/defect size to critical size for one lifetime

   Critical damage size greater than two times NDT size

   No delamination growth.

- Soft Skin
  Basic Skin 19 – 21 Plies
  (5/84/11) to (5/85/10)
- Planking Up to 47-ply
  (51/43/6)
- I-Shaped Stringer



F94-HPK/02

FIGURE 2. UPPER WING SKIN/STRINGER CROSS SECTION OF THE SMALL AIRPLANE.

SECTION 3

DAMAGE SCENARIOS

AC 25.571-1A provides guidelines to establish appropriate criteria for damage tolerance design of aircraft structures. Under these guidelines the extent of damage is established in relation to inspectability and damage extension characteristics of metal structures. For tension dominated metal structures the damage scenarios resulting from these guidelines provide a sound basis for damage tolerance design. Even though the type of damage based on AC 25.571-1A is basically crack-like damage, reference 3 adopted this approach for damage tolerance design of composite fuselage structures. The structural integrity for the tension-dominated components designed with this approach is believed to be adequate. However, the damage tolerance capability of the compression-dominated components is not properly addressed using this approach. This is because the sensitivity and severity to damage type are significantly different between tension components and compression components.

Sources of in-service damage to composite aircraft structures are reviewed in references 11, 13 and 14. Based on the in-service composite structural maintenance records, the type of damage may be summarized as follows:

TABLE 1. DISTRIBUTION OF IN-SERVICE DAMAGE ON COMPOSITE STRUCTURES.

TYPE OF DAMAGE	OCCURRENCE	PERCENT
IMPACT RELATED	95	55.9
DISBOND, SEPARATION, AND DELAMINATION	30	17.6
HANDLING AND OPERATION RELATED	21	12.4
LIGHTENING, FIRE RELATED	15	8.8
REPAIR, PRODUCTION, AND ENVIRONMENT	9	5.3
TOTAL	170	100.0

The results of these surveys indicate that the most common type of damage that occurs in composite structures is impact related damage. This type of damage may be caused by a variety of sources ranging form tool drop during routine service to impact with ground handling equipment. The extent of damage may range from barely visible dents to through penetration holes to gouges or torn skin.

These results show that crack-like damage is not a serious threat to in-service composite structures. They also show that a multitude of damage scenarios must be considered for damage tolerance of composites. Damage scenarios must be developed based on realistic damage type, the extent of damage, and the source of damage that potentially threaten the integrity of the composite structure throughout its service life.

In relation to the damage severity, a comprehensive composite material defect/damage sensitivity assessment was conducted in reference 9. This assessment was conducted based on the results of a number of government sponsored research programs and service experience over a period of years. The assessment concluded that impact is the most severe defect/damage type for compressively loaded structures. Compression strength was selected for the defect/damage comparison because it is generally the most critical loading mode for damaged composite structures. The results of reference 9 and other studies are integrated into a recommendation in AC-107A (reference 15) which states that, "It should be shown that impact damage that can be realistically expected from manufacturing and service, but not more than the established threshold of detectability for the selected inspection procedure, will not reduce the structural strength below ultimate load capability."

Damage scenarios for damage tolerance consideration in military aircraft have been established based on a number of Air Force and Navy funded programs. Guidance for initial flaw/damage assumptions is given in references 9 and 16 as follows:

TABLE 2. FLAW/DAMAGE ASSUMPTIONS FOR COMPOSITE STRUCTURES.

FLAW/DAMAGE TYPE	FLAW/DAMAGE SIZE
SCRATCHES	SURFACE SCRATCH 4.0 INCHES IN LENGTH AND 0.02 INCH IN DEPTH
DELAMINATION	INTERPLY DELAMINATION EQUIVALENT TO A 2.0-INCH-DIAMETER CIRCLE WITH DIMENSIONS MOST CRITICAL TO ITS LOCATION
IMPACT DAMAGE	DAMAGE FROM A 1.0-INCH-DIAMETER HEMISPHERICAL IMPACTOR WITH 100-FT-LBS OF KINETIC ENERGY OR WITH THAT KINETIC ENERGY REQUIRED TO CAUSE A DENT 0.10 INCH DEEP, WHICHEVER IS LESS

The damage scenarios discussed earlier in this section emphasized impact damage, delaminations and cracks. Even though the surveys of references 13 and 14 indicate that disbond is the second most frequently observed in-service damage, requirements or guidelines for this type of defect/damage are not available. As pointed out in reference 10, the integrity of bonded structures is of concern to the FAA because there is no satisfactory nondestructive inspection technique currently available to reliably detect understrength bonds. Manufacturers are currently required to assess each bonded structure, critical to safe flight, and determine the maximum disbond size that can exist consistent with the capability of the remaining structure to sustain limit load. Disbonds greater than these must be prevented by design features if there is a finite possibility that these disbonds might grow to catastrophic sizes before detection.

In the present study, three competing damage types are considered. They are: impact damage, delaminations, and skin/stiffener disbonds. The influence of these damage types on the residual strength of the two composite structures discussed in section 2 will be analytically determined. The impact threat used in the original design of these structures will be used as the baseline damage scenario. Critical disbond size will be determined so that the residual strength of the damaged structure is equivalent to that of the structure with the baseline impact damage. Finally, the severity of interply delaminations will be compared to that of the impact damaged structure.

SECTION 4

ANALYSIS OF DAMAGED STRUCTURES

Analyses were conducted on the two composite structures described in section 2 containing damage which was discussed in section 3. The strength analysis methods selected were based on the expected failure modes associated with each damage scenario. A schematic of the competing failure modes as a function of the damage size is shown in figure 3. In this figure, the baseline strength is assumed to be the impact damage tolerance design strength for the respective structure. This strength will remain as a constant. The residual strength of the structure with a skin/stiffener disbond is a function of the disbond length. It may be noted that a through-the- width (of the stiffener) disbond is used throughout the present study. The failure mode for the structure with a disbond under compression loading is most likely a stability related failure. This is because out-of-plane deformation of the skin or the stiffener caused by buckling can induce disbond growth and further reduce the strength. The buckling mode can be local skin or stiffener buckling or global panel buckling. In figure 3, the lower bound buckling strength is shown. The objective of the analysis is then to determine the maximum disbond length so that buckling (local or global) will not occur below the baseline strength of the impact damaged structure.

Interply delamination is the other damage type considered in this study. Because delamination can take place at any ply interface, the lower bound delamination strength is shown in figure 3. A conservative approach is adopted in the delamination analysis; that is, no delamination growth is allowed throughout the service life of the structure. This is conservatively equivalent to no delamination growth under static compression loading. The delamination analysis method developed in reference 9 is used in this study.

In addition to the residual strength of the damaged structure, figure 3 also shows the material compression strength of the undamaged structure for reference purposes.

Details of the analysis for the two structures are discussed in the following paragraphs.

4.1 LARGE-AIRPLANE WING STRUCTURE

The impact damage scenario adopted for the wing box test in reference 9 is shown in figure 4. All impacts were on the box upper surface panel, which was loaded primarily in compression. Impacts were in pairs, consisting of 100 ft-lb and an adjacent 20 ft-lb. This damage approach is analogous to the damage tolerance scenario for the metals, where allowance is made for the possibility of continuing crack growth at adjacent fastener holes.

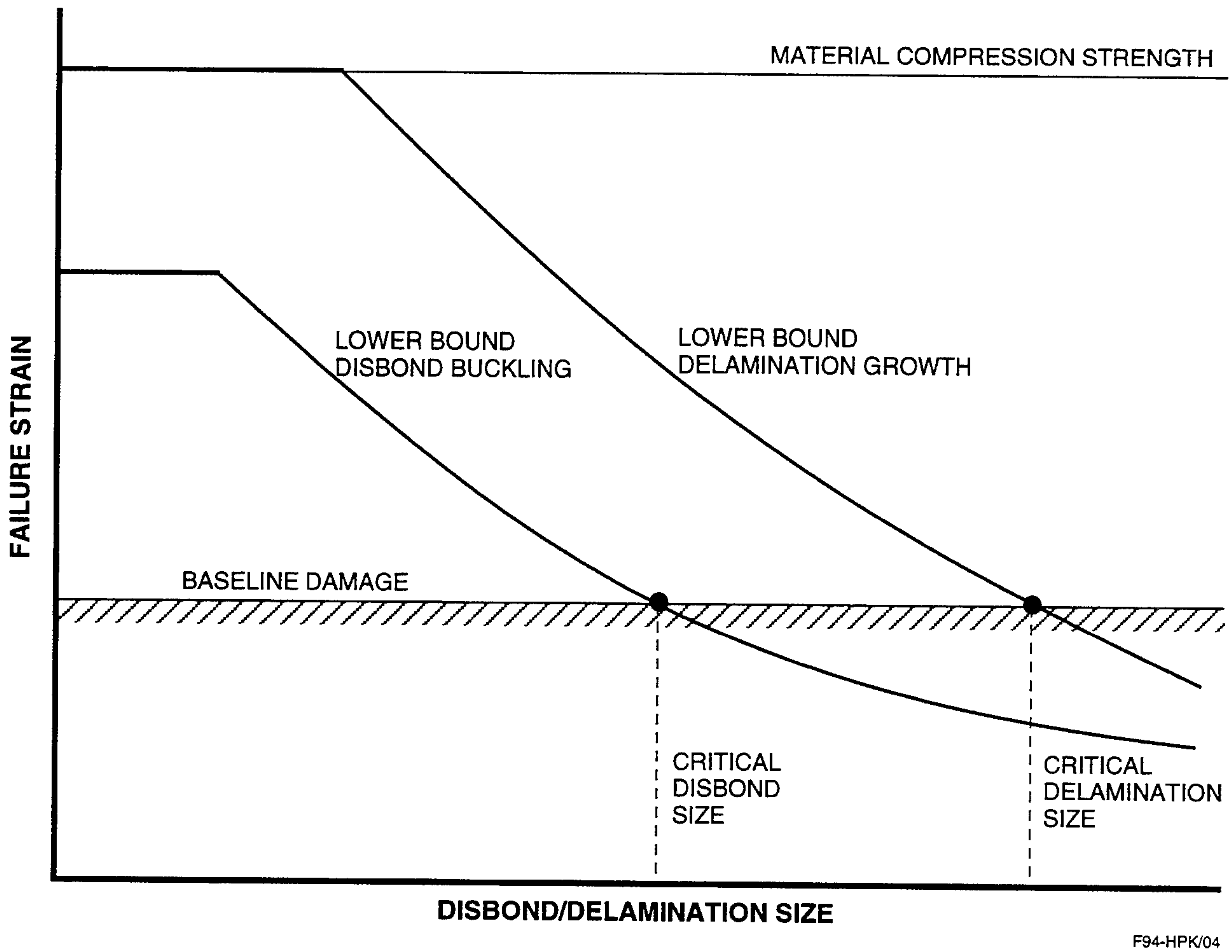
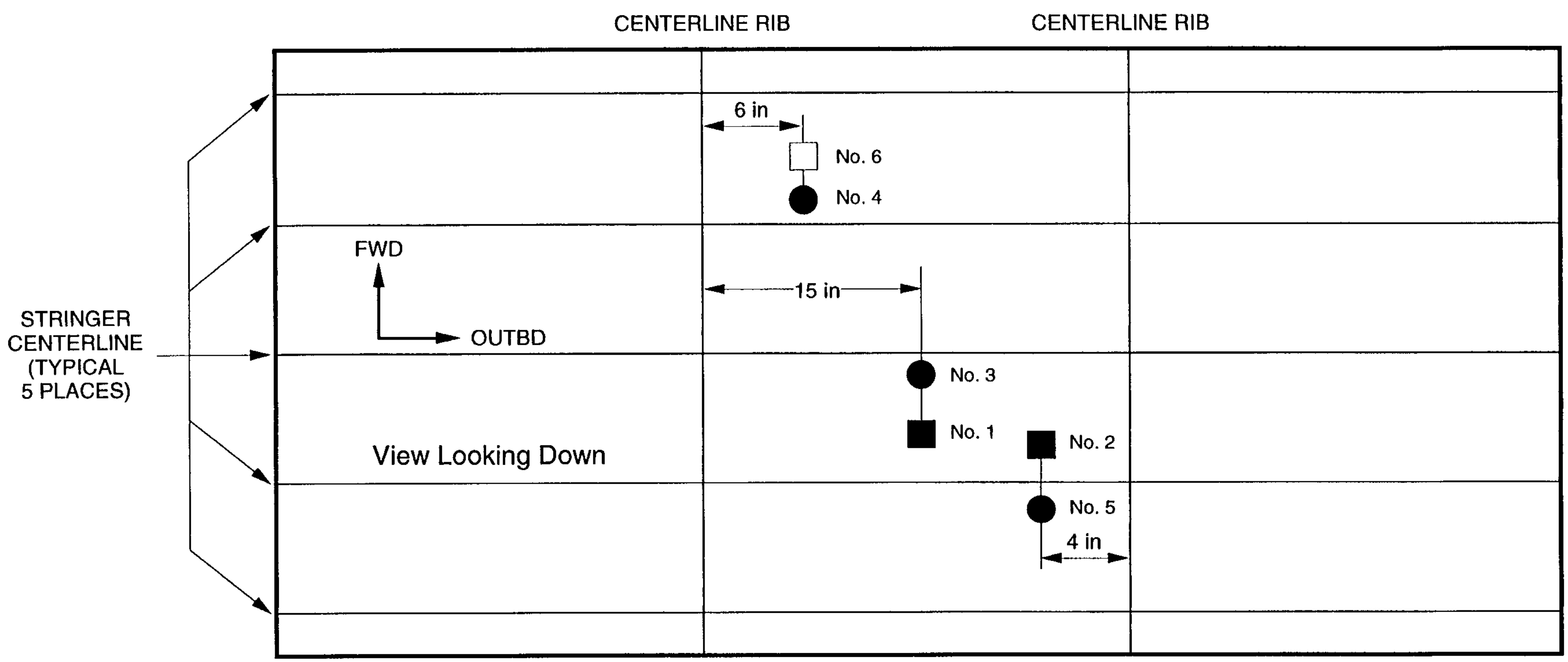


FIGURE 3. SCHEMATIC OF DAMAGED STRUCTURE FAILURE STRAIN.



NOTE: All Five Initial Impacts Are Made at the Edge of the Stringer Pads.

LEGEND:

- ● INITIAL 100 ft-lb DAMAGE
- ■ INITIAL 20 ft-lb DAMAGE
- □ 20 ft-lb DAMAGE IMPOSED AFTER 200,000 CYCLES (ONE LIFETIME)

F94-HPK/05

FIGURE 4. LARGE-AIRPLANE WING BOX IMPACT DAMAGE LEVELS AND LOCATIONS.

Two spectrum loading lifetimes were applied to the box. The wing box was thoroughly inspected after completing the cyclic loading. The box was then loaded statically to failure. Box failure was at approximately 105 percent of limit load and the axial strain in the failure region was resolved into a principal strain of 0.0042. This strain value is used as the baseline strength for the damaged structure.

The structural details of the upper surface panel were shown in figure 1. A panel 48 inches wide and 35.4 inches long was used in the analysis. The basic soft skin is a 36-ply, (10/80/10) layup with a stacking sequence as

$$(\pm45/90/\pm45/\pm45/0/\pm45/90/\pm45/\pm45/0/\pm45)_S$$

The plank under each stringer is 3.9 inches wide with 19 additional 0-degree plies interleaved into the basic skin. The layup in this region then becomes 55-ply, (42/51/7). The 0-degree plies are properly dropped off to 4.3 inches width before the basic skin is resumed. A 3-ply, (0/90/0) precured strip is placed between the plank and the stringer. The I-shaped stringers are 1.87 inches high, with a 2.78-inch-wide flange and 1.88-inch-wide cap. The layup for the stringers is approximately (60/30/10), with a 12-ply-thick flange and 24-ply-thick web and cap. The stringers are spaced 8 inches between centerlines. A detailed cross-sectional view of the skin/stringer arrangement is shown in figure 5.

The mechanical properties for the cross section were computed based on classical lamination theory for the AS4/3501-6 graphite/epoxy material system. They are listed below:

TABLE 3. MECHANICAL PROPERTIES FOR THE LARGE-AIRPLANE WING STRUCTURE.

	Ex (msi)	Ey (msi)	Gxy (msi)	Vxy	Vyx
BASIC SKIN	5.312	5.312	4.008	0.516	0.516
PLANK	9.964	4.779	2.917	0.489	0.235
PLANK+STRIP	10.192	4.946	2.810	0.460	0.223
TRANSITION*	6.454	5.226	3.728	0.503	0.407
STRINGER	12.506	4.523	2.203	0.402	0.145

* The transition zone is 0.4 in. wide and the properties are weighted averages.

Buckling analysis based on equivalent sectional properties was first conducted for the undamaged panel. Euler buckling, panel buckling as well as local skin buckling were considered. Simply supported boundary conditions were used in the panel and local skin analysis, using the method of reference 17. Two models were used for the local skin buckling analysis. The first model considered was the basic skin only, which resulted in a panel 4.1 inches wide and 35.4

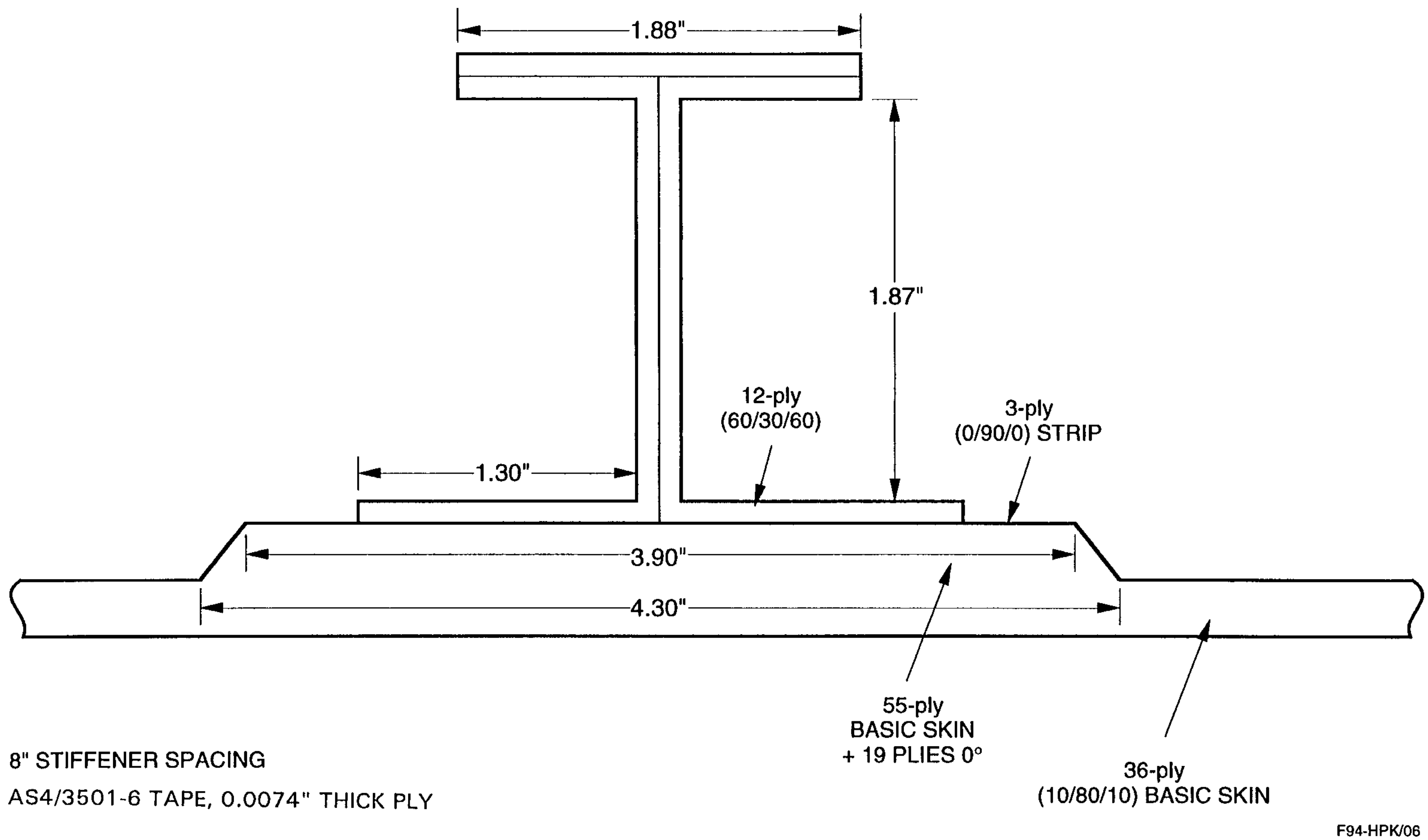


FIGURE 5. STRINGER DETAILS FOR THE LARGE-AIRPLANE WING.

inches long. The second model considered a full-skin bay with the skin and plank combination for a panel 8 inches wide by 35.4 inches long. The buckling strain in comparison with the baseline damaged structural strength is given below.

TABLE 4. BUCKLING STRAINS FOR THE LARGE-AIRPLANE WING.

FAILURE MODE	FAILURE STRAIN	STRAIN RATIO
COMPRESSION	0.0110	2.619
EULER BUCKLING	0.0051	1.203
PANEL BUCKLING	0.0056	1.338
LOCAL SKIN BUCKLING	0.0249	5.922
LOCAL BAY BUCKLING	0.0061	1.455
BASELINE DAMAGE	0.0042	1.000

The results above indicate that the baseline impact damage controls the compression strength of the upper skin panel. Buckling strength (local or global) of the undamaged panel is high in comparison to the residual strength of the panel with baseline impact damage. For comparison purposes, the compression failure strength is also listed above.

For the skin/stringer disbond type of damage, the first scenario assumed is a complete disbond between the skin and the stringer. Under this assumption, the disbonded stringer would respond to the applied compression load as an independent structural unit. Global and local buckling analyses were conducted using the same method discussed earlier. For the global buckling, the equivalent panel properties were computed with the absence of one stringer. The axial stiffness of the damaged panel is reduced by 6.2 percent as compared to the undamaged panel. This resulted in a 10.9 percent reduction in the Euler buckling strain. The axial bending rigidity of the panel was reduced by 16.6 percent because of the assumed damage. The panel buckling strain reduction caused by the damage is 18.1 percent. However, both the Euler buckling and panel strains, 0.0045 and 0.0046, respectively, are higher than the failure strain of the structure with baseline damage. Therefore, global buckling is not the critical failure mode for the damaged structure.

The local skin buckling strain is relatively high for the undamaged structure. This strength is not significantly reduced by the disbond because the buckling analysis of the undamaged configuration for this failure mode already assumed that the skin reacts to applied load independent of the remainder of the structure.

The local bay buckling behavior is significantly affected by the disbond. This is because in absence of one stringer, two skin bays would react to the applied load as a single independent local unit. As a result, a 16-inch-wide, 35.4-inch-long panel with the skin and plank combination was used in the analytical model. The buckling strain for such a model is 0.0016, which is significantly lower than the baseline strain of 0.0042.

A less conservative model considering only the middle section of the disbonded bay was also used to further evaluate the strength of this damage configuration. This model includes the plank area under the disbonded stringer and the adjacent basic skin but excluding the plank areas under the intact stringers. The reason for using this model is that the plank areas under the intact stringers are likely to deform with the intact stringers rather than to react as a part of the independent structural unit. This results in a 12.1-inch-wide, 35.4-inch-long panel. The local buckling strain for such a panel is 0.0031, which is 73.1 percent of the baseline strength. This model will be referred to as Local Model II, and the more conservative model discussed previously as Local Model I.

The buckling strength of the upper skin panel with one stringer completely disbonded is summarized below.

TABLE 5. BUCKLING STRAINS FOR THE LARGE-AIRPLANE WING WITH ONE STRINGER COMPLETELY DISBONDED.

FAILURE MODE	FAILURE STRAIN	STRAIN RATIO
BASELINE DAMAGE	0.0042	1.000
EULER BUCKLING	0.0045	1.071
PANEL BUCKLING	0.0046	1.096
LOCAL BAY BUCKLING*	0.0016	0.386
LOCAL BAY BUCKLING**	0.0031	0.731

* Local Model I, skin plus planks under disbonded stringer and adjacent intact stringers.

**Local Model II, skin plus plank under disbonded stringer but excluding planks under adjacent intact stringers.

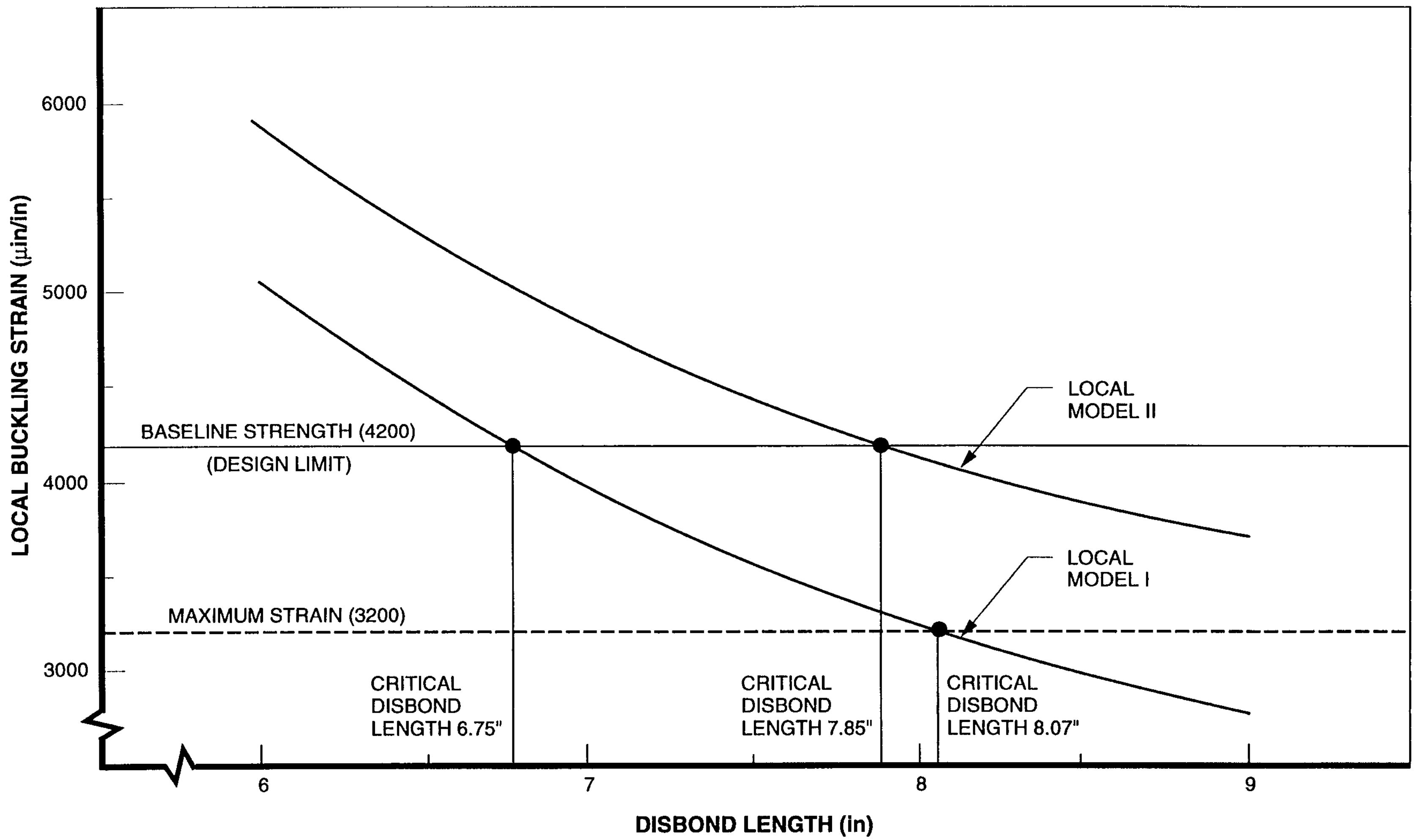
The buckling analysis results shown above indicated that a stringer completely disbonded is a more severe damage scenario than the baseline impact damage This type of damage, if not detected and repaired, will significantly degrade the integrity of the structure. Design features must be provided to prevent this damage from occurring throughout the service life of the structure. One approach is to design the structure using a complete stringer disbond as one of the damage tolerance criteria. This would impose a significant weight penalty on the structure. An alternative approach would be to establish a critical disbond length and size the structure based on the impact damage tolerance criterion, which is more familiar to the current structural designers and analysts. The certification approach will be discussed in section 5. The critical disbond length is discussed in the following paragraphs.

The local buckling strain as a function of disbond length was computed using the plate buckling analysis method of reference 17 with simply supported boundary conditions. The results are shown in figure 6. The figure shows results for both Local Models I and II. The buckling strengths are compared to the baseline strain of 0.0042 to determine the critical disbond length. For Local Model I, the critical disbond length is 6.75 inches. The critical disbond length becomes 7.85 inches when the less conservative model, Local Model II, is used in the analysis.

A different approach, based on the equivalent total panel failure load, was also used to determine the critical disbond length. The wing panel in reference 9 was designed for an axial compression load of 25,000 lb/in. at ultimate condition. This is equivalent to a total load of 1,200,000 lbs for the 48-inch-wide panel. At limit condition the panel load is then 800,000 lbs. Applying the same approach as in reference 9 and using a factor of 1.05 for allowance of biaxial effects, the baseline total load requirement is 840,000 lbs at limit. The equivalent panel load for local buckling as a function of disbond length is shown in figure 7. The figure shows the results for both local analysis models. The critical disbond lengths determined by this approach are slightly shorter than the strain approach. The critical disbond length is 6.15 inches for Local Model I and 6.90 inches for Local Model II.

Interply delaminations in the basic skin is another damage scenario investigated in this study. For this type of damage the damage tolerance design criterion considered is no delamination growth throughout the service life of the structure. The delamination analysis method developed in reference 9 was used to evaluate the criticality of this type of damage. Because of the no growth criterion, only the delamination buckling analysis was considered.

In the delamination analysis, a circular delamination was assumed. The minimum delamination size at which local buckling failure of the delaminated region can occur was first determined. The buckling strain corresponding to this delamination size was then computed. Finally, the equivalent panel failure load was obtained. The buckling strain was compared to the baseline damaged structural strain of 0.0042 to find the strain ratio. The equivalent panel failure load was compared to the design limit load of 840,000 lbs to find the load ratio. The results, in terms of delamination depth are shown in the following table. These results are based on a delamination quality coefficient of 0.33, which is the average value determined in references 9 and 12. The concept of delamination boundary quality is discussed in references 9 and 12. A delamination quality coefficient, ranging from 0.0 to 1.0, is used in these references to quantify



F94-HPK/07

FIGURE 6. CRITICAL DISBOND LENGTH BASED ON STRAIN, LARGE-AIRPLANE WING.

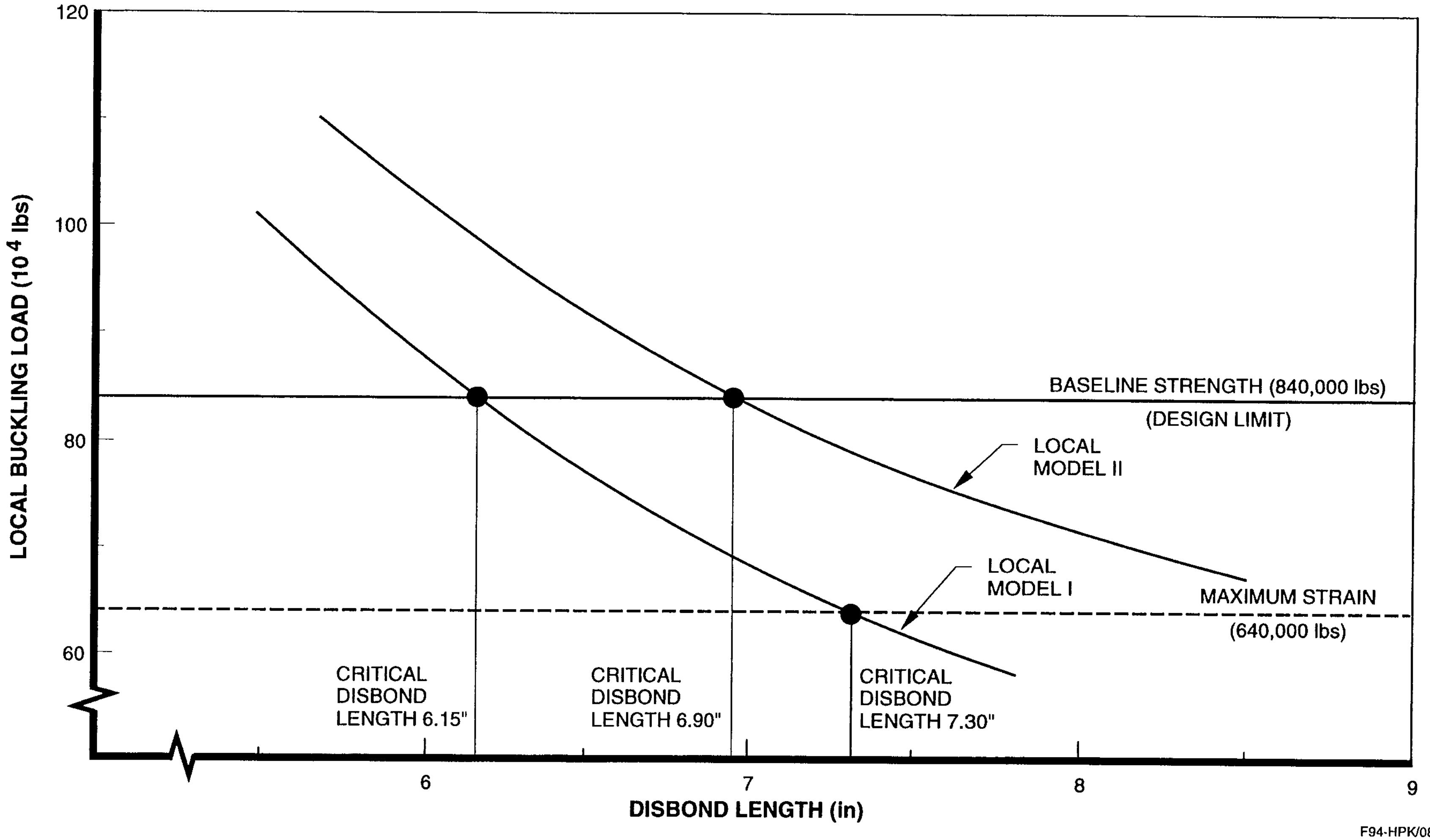


FIGURE 7. CRITICAL DISBOND LENGTH BASED ON LOAD, LARGE-AIRPLANE WING.

the boundary quality. In relation to the delamination analysis, a delamination quality coefficient of 0.0 corresponds to a fully clamped boundary and 1.0 corresponds to a simply supported delamination boundary.

TABLE 6. CRITICAL DELAMINATION SIZES FOR THE LARGE-AIRPLANE WING SKIN.

NO. OF PLIES DELAMINATED	CRITICAL DELAMINATION DIAMETER (in.)	FAILURE STRAIN	STRAIN RATIO	PANEL LOAD (kips)	LOAD RATIO
1	8.88	0.0108	2.575	1,994	2.374
2	8.58	0.0106	2.530	1,960	2.333
3	8.35	0.0103	2.443	1,892	2.252
4	8.04	0.0101	2.402	1,860	2.214
5	7.74	0.0099	2.364	1,830	2.179
6	7.44	0.0098	2.328	1,803	2.147
7	7.14	0.0097	2.298	1,779	2.118
8	7.19	0.0086	2.055	1,592	1.895
9	6.89	0.0086	2.037	1,578	1.878
10	6.58	0.0085	2.029	1,571	1.871
11	6.36	0.0083	1.984	1,536	1.829
12	6.06	0.0084	2.008	1,555	1.852
13	5.75	0.0086	2.056	1,593	1.896
14	5.45	0.0090	2.136	1,655	1.970
15	5.15	0.0095	2.258	1,749	2.082
16	5.21	0.0089	2.113	1,636	1.948
17	4.90	0.0098	2.323	1,799	2.142
18	4.59	0.0110	2.619	2,028	2.415

These results indicate that the critical delamination size decreases with the delamination depth. The absolute minimum delamination diameter is 4.59 inches at the laminate midplane (18-ply deep). The lowest strain for delamination failure is 0.0083 for an 11-ply-deep delamination. The corresponding strain factor is 1.984. The delamination strength is affected by the laminate staking sequence. The variation of the delamination strain as a function of the delamination depth is shown in figure 8.

The overall strain and load ratios are very high. This indicates that delamination is a less severe threat as compared to impact or disbond. Thus, damage tolerance design for impact or skin/stiffener disbond should, in general, account for the strength reduction due to interply delamination. Additional damage tolerance criteria for delamination are not needed.

Strength requirement may be less than limit load if the effect of the damage is such that it is noticeable to the pilot in terms of performance.

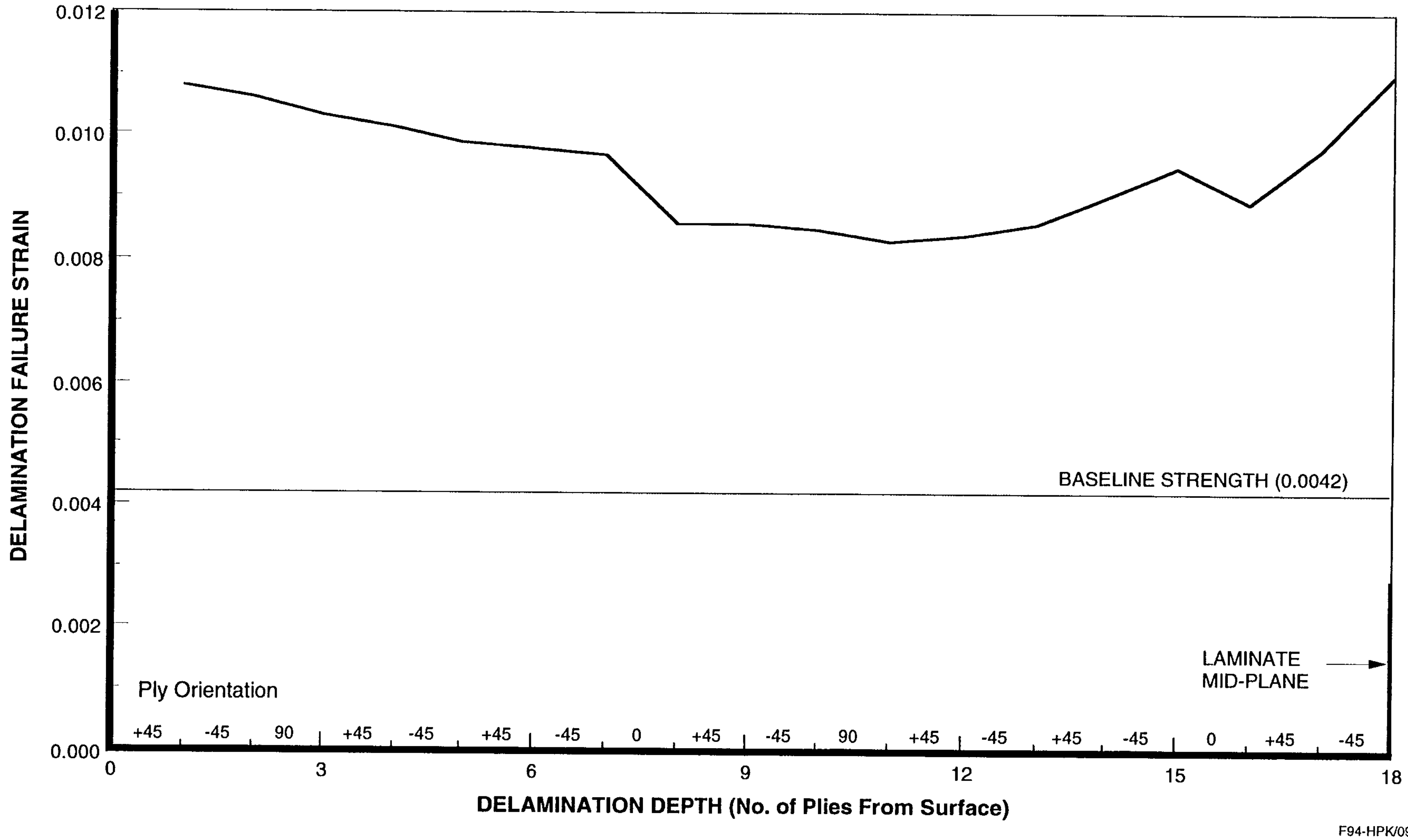


FIGURE 8. DELAMINATION STRENGTH AS A FUNCTION OF DELAMINATION DEPTH, LARGE-AIRPLANE WING.

4.2 SMALL-AIRPLANE WING STRUCTURE

The impact damage tolerance design allowable used in the small-airplane wing structure was 0.00365. The maximum compression strain in the upper wing skin was 0.00362 under ultimate conditions. Based on the design allowable strain and adopting the 1.05 factor, as in the large airplane wing, to provide an allowance for biaxial effects, the baseline damaged structural strength used in the current study is 0.00385. Two locations in the upper wing skin were selected for this study. The first is a typical section with a 21-ply basic soft skin and 7.5-inch stringer spacing. The details of the cross section is shown in figure 9. A 5-stringer, 45-inch-wide and 31-inch-long panel is used in the analysis. These dimensions are typical of the actual wing skin. The basic skin is of (5/86/9) layup with a stacking sequence of

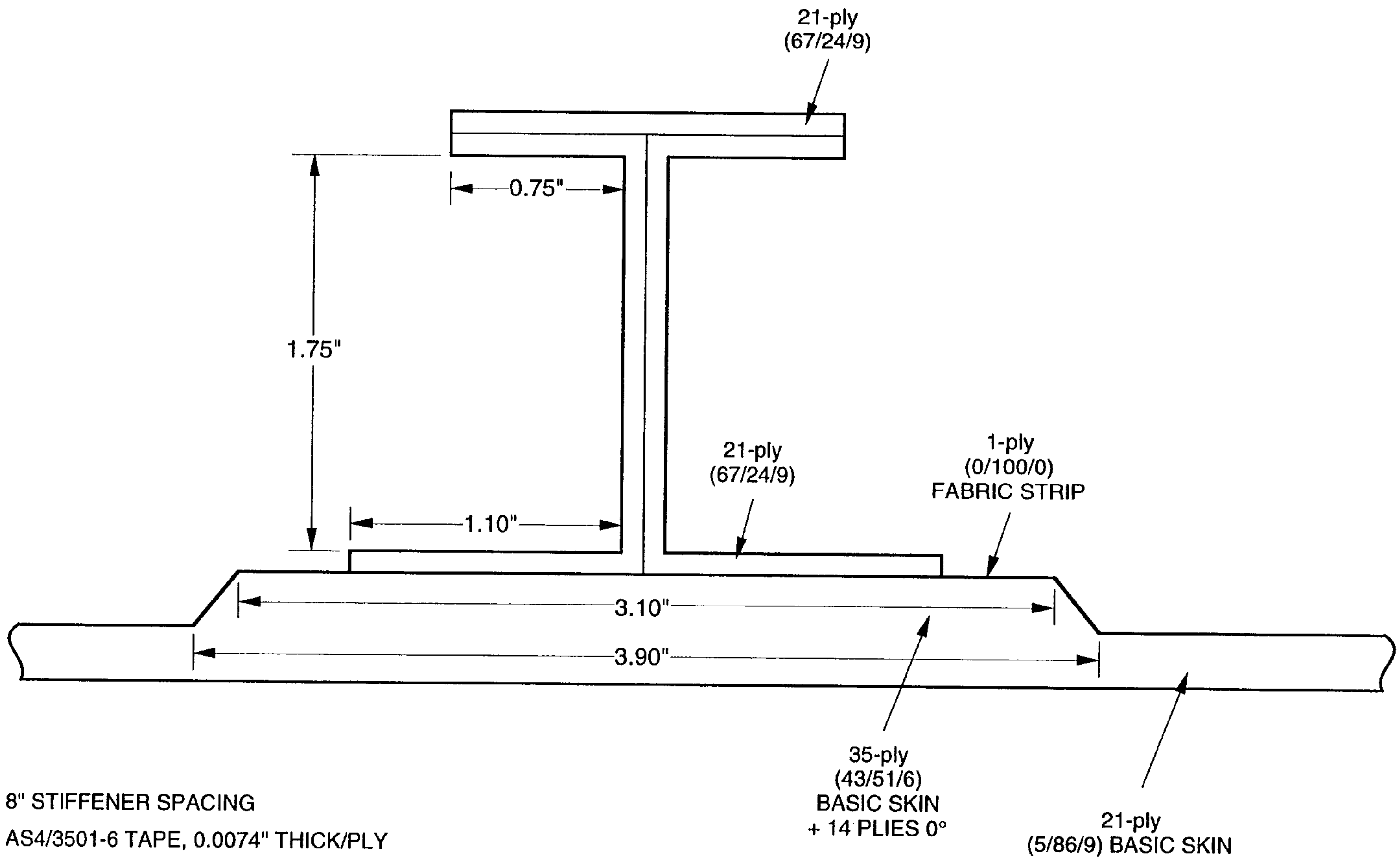
$$(+45/90/-45/\pm45/\pm45/\pm45/+45/0/\pm45/\pm45/\pm45/\pm45/90/+45)t.$$

The plank area under the stringer is 35 plies thick with fourteen 0-degree plies interleaved into the basic skin to form a (43/51/6) laminate. The plank is 3.1 inches wide on the stringer side and the 0-degree plies are properly dropped off to the basic skin. The plank width on the skin side is 3.9 inches. One ply of fabric strip is placed between the plank and the stringer. The stringer flange near the skin is 1.1 inches wide on each side of the I-shaped section and 0.75 inches wide away from the skin. The stringer, made from channel sections, has a layup of (67/24/9) and is 21 plies thick. A 21-ply-thick laminate with the same layup is used for the cap. The stringer is 1.75 inches high. The material for the panel is IM6/3501-6 carbon/epoxy with a ply thickness of 0.0074 inch, except for the fabric strip, which is of AS4/3501-6. The mechanical properties for the cross section are computed and are listed as

TABLE 7. MECHANICAL PROPERTIES FOR THE SMALL-AIRPLANE WING STRUCTURE--TYPICAL SECTION.

	Ex (msi)	Ey (msi)	Gxy (msi)	Vxy	Vyx
BASIC SKIN	4.665	5.296	4.762	0.546	0. 613
PLANK	11.218	4.691	3.237	0.519	0.217
STRIP	2.511	2.511	4.519	0.744	0.744
TRANSITION*	6.303	5.145	4.381	0.535	0.437
STRINGER	15.345	4.390	2.009	0.335	0.096

* the transition zone is 0.4-inch-wide and the properties are weighted averages



8" STIFFENER SPACING

AS4/3501-6 TAPE, 0.0074" THICK/PLY

F94-HPK/10

FIGURE 9. STRINGER DETAILS FOR THE TYPICAL SMALL-AIRPLANE WING SECTION.

The second section selected for analysis is a typical section in the inboard area of the wing structure. This section is similar to the section described above. The basic skin is 19 plies thick with a layup of (5/84/11) and a stacking sequence of

(+45/90/-45/±45/±45/±45/0/-45/±45/±45/±45/90/+45)t.

The plank area is 39 plies thick with 20 additional 0-degree plies interleaved into the basic skin to form a laminate of (54/41/5) layup. The stringer, made from channel sections, is 22 plies thick with (68/23/9) layup. The stringers are 1.8 inches high. The stringer spacing is also 7.5 inches. The panel dimensions used in the analysis are also 45 inches wide and 31 inches long. The mechanical properties for this section are as follows:

TABLE 8. MECHANICAL PROPERTIES FOR THE SMALL-AIRPLANE WING STRUCTURE--INBOARD SECTION.

	Ex (msi)	Ey (msi)	Gxy (msi)	Vxy	Vyx
BASIC SKIN	4.786	5.500	4.695	0.524	0. 602
PLANK	13.121	4.238	2.774	0.496	0.160
STRIP	2.511	2.511	4.519	0.744	0.744
TRANSITION*	7.660	5.065	4.033	0.514	0.340
STRINGER	15.602	4.256	1.961	0.335	0.091

* the transition zone is 0.4 inch wide and the properties are weighted averages

Global and local buckling analyses were conducted for the undamaged structure. Analyses were conducted for the upper skin at a typical section and at an inboard section. A 45-inch-wide panel with 5 stringers spaced at 7.5 inches was used in the global analysis for both locations. Local skin buckling analysis used a 3.6-inch wide simply supported panel consisting of basic skin only. The local bay buckling considered a 7.5-inch-wide panel consisting of skin and plank area for the bay with equivalent properties. The panel length used in all the analyses was 31 inches. The results of the buckling analyses are as follows:

TABLE 9. BUCKLING STRAINS FOR THE SMALL-AIRPLANE WING SECTIONS.

FAILURE MODE	FAILURE STRAIN	STRAIN RATIO
TYPICAL SECTION		
BASELINE DAMAGE	0.00385	1.000
EULER BUCKLING	0.00843	2.304
PANEL BUCKLING	0.00978	2.541
LOCAL SKIN BUCKLING	0.01395	3.623
LOCAL BAY BUCKLING	0.00269	0.700
INBOARD SECTION		
BASELINE DAMAGE	0.00385	1.000
EULER BUCKLING	0.01068	2.774
PANEL BUCKLING	0.01232	3.199
LOCAL BAY BUCKLING	0.01106	2.873
LOCAL BAY BUCKLING	0.00242	0.629

These results indicate that for the undamaged structure, local bay buckling is the critical failure mode. This is believed to result from one of the design criteria, which allows skin, spar, and rib webs to buckle beyond limit load (see section 2.2). However, this mode of buckling is not likely to occur for the undamaged structure, because of the constraint of the stringer on the skin bay, which is neglected in the analytical model. The strain ratios for other failure modes are relatively high, indicating that the impact damage tolerance requirement is the primary design driver.

Analyses were also conducted for the damage scenario of one stringer completely disbonded from the skin panel. As in the case of the large airplane wing, global and local buckling analyses were performed for the damaged structure. The results of these analyses are as follows:

TABLE 10. BUCKLING STRAINS FOR THE SMALL-AIRPLANE WING SECTIONS WITH ONE STRINGER COMPLETELY DISBONDED.

FAILURE MODE	FAILURE STRAIN	STRAIN RATIO
TYPICAL SECTION		
BASELINE DAMAGE	0.00385	1.000
EULER BUCKLING	0.00759	1.971
PANEL BUCKLING	0.00892	2.316
LOCAL BAY BUCKLING*	0.00069	0.179
LOCAL BAY BUCKLING**	0.00105	0.273
INBOARD SECTION		
BASELINE DAMAGE	0.00385	1.000
EULER BUCKLING	0.00913	2.372
PANEL BUCKLING	0.00987	2.562
LOCAL BAY BUCKLING*	0.00066	0.172
LOCAL BAY BUCKLING**	0.00084	0.218

* Local Model I, skin plus planks under disbonded stringer and adjacent intact stringers

**Local Model II, skin plus plank under disbonded stringer but excluding planks under adjacent intact stringers.

These results show that the global buckling strength is not significantly degraded by the damage for both sections analyzed. The typical section has a 8.9 percent reduction in panel buckling strain and the reduction is 19.9 percent for the inboard section. But the failure strains for both sections remain very high as compared to the baseline impact damage strength.

The local buckling strength for panel with one stringer completely disbonded, on the contrary, is significantly reduced for both sections considered. Two local models, as described earlier for the large airplane structure, were used in the analysis. Local Model I considered a 15-inch-wide by 31-inch-long panel. The strain ratio is 0.179 for the typical section and 0.172 for the inboard section. Local Model II used a 11.9-inch-wide by 31-inch-long panel. The results show a strain ratio of 0.273 for the typical section and 0.218 for the inboard section. These results, similar to the large airplane wing, indicate that a completely disbonded stringer is a more severe threat to the integrity of the structure, as compared to the baseline impact damage.

Similar to the large airplane wing, the structure designed for the baseline impact damage does not provide a sufficient margin of safety for the complete stringer disbond type of damage.

In order to assure the damage tolerance capability of the structure, the critical disbond length must be specified. The critical disbond lengths for the two sections considered were determined using the same analysis method as that for the large airplane wing. The failure strength as a function of the disbond length for the typical section is shown in figure 10. The figure shows that results obtained for the two local models are not significantly different. The critical disbond length obtained using Local Model I is 5.04 inches, and it is 5.08 inches when Local Model II is used in the analysis. Similar results for the inboard section are shown in figure 11, where the critical disbond lengths are 5.50 inches and 5.27 inches for Models I and II, respectively. These results were obtained based on the residual strength requirement of limit load (3850/1.5 = 2567).

Strength reduction due to interply delamination in the basic skin was determined based on the analysis method of reference 9. As in the case of the large airplane wing, a circular delamination was assumed. The critical delamination size for delamination growth, the corresponding failure strain and the strain ratio were computed. The results for the typical section are shown in the following table.

TABLE 11. CRITICAL DELAMINATION SIZES FOR SMALL-AIRPLANE WING SKIN--TYPICAL SECTION.

NO. OF PLIES DELAMINATED	CRITICAL DELAMINATION DIAMETER (in)	FAILURE STRAIN	PANEL LOAD (kips)	STRAIN/ LOAD RATIO
1	6.25	0.00858	1,580	2.229
2	6.03	0.00789	1,453	2.050
3	5.67	0.00761	1,402	1.977
4	5.32	0.00737	1,356	1.913
5	4.96	0.00718	1,321	1.864
6	4.61	0.00708	1,304	1.840
7	4.25	0.00714	1,315	1.854
8	3.90	0.00744	1,371	1.933
9	3.55	0.00814	1,4981	2.113
10	3.21	0.00947	1,743	2.459

As in the large airplane wing, these results again show that skin delamination is a much less severe damage threat to the structural integrity. The strain or load ratio is larger than 1.80. It may be noted that the baseline damaged panel load was computed based on the impact damage design allowable strain of 0.00385. The baseline panel failure load is 709 kips. The delamination strength as a function of delamination depth is shown in figure 12. As can be seen from the figure, the residual strength due to delamination is significantly higher than the baseline impact damage strength.

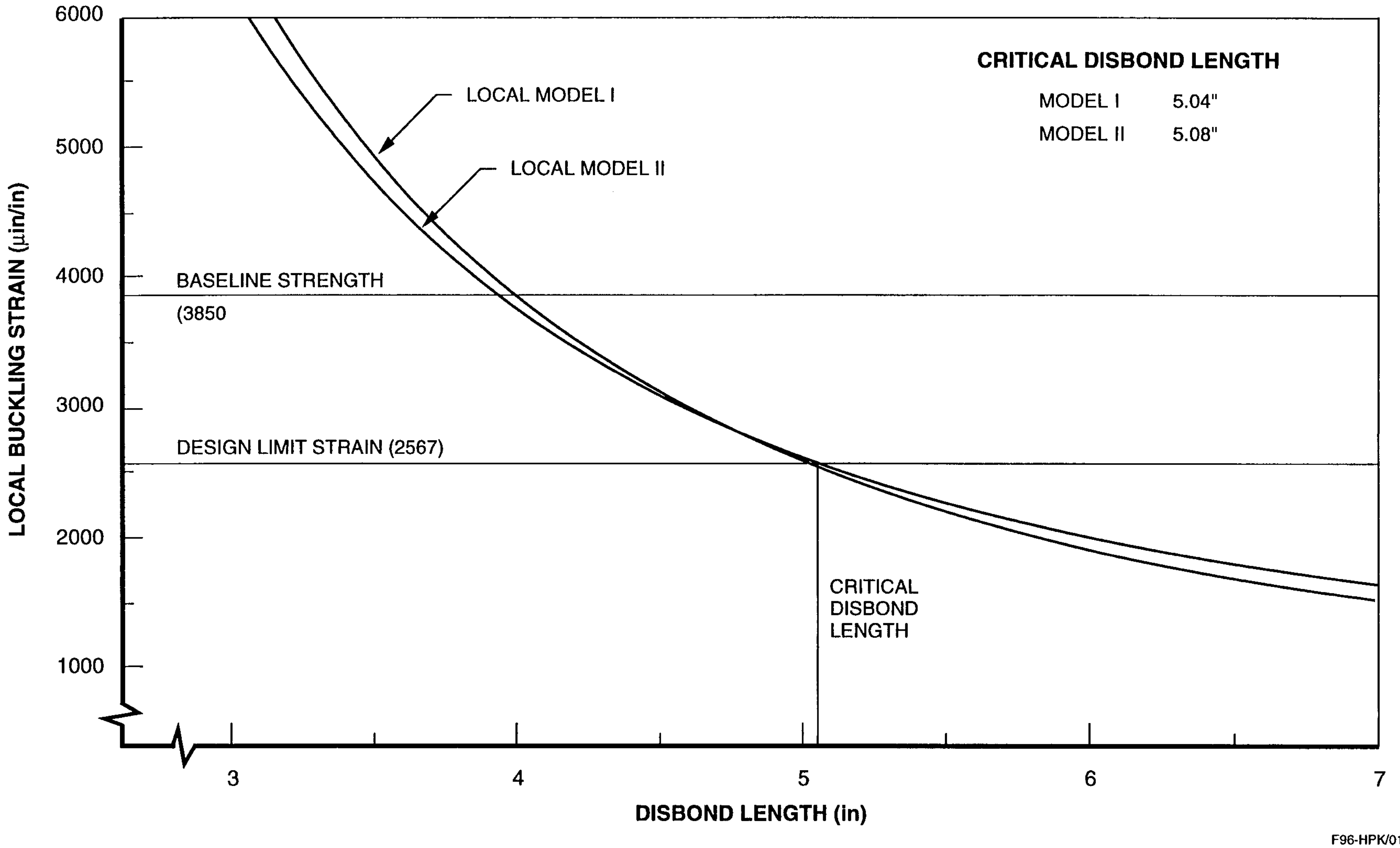


FIGURE 10. CRITICAL DISBOND LENGTH FOR THE TYPICAL SECTION, SMALL-AIRPLANE WING.

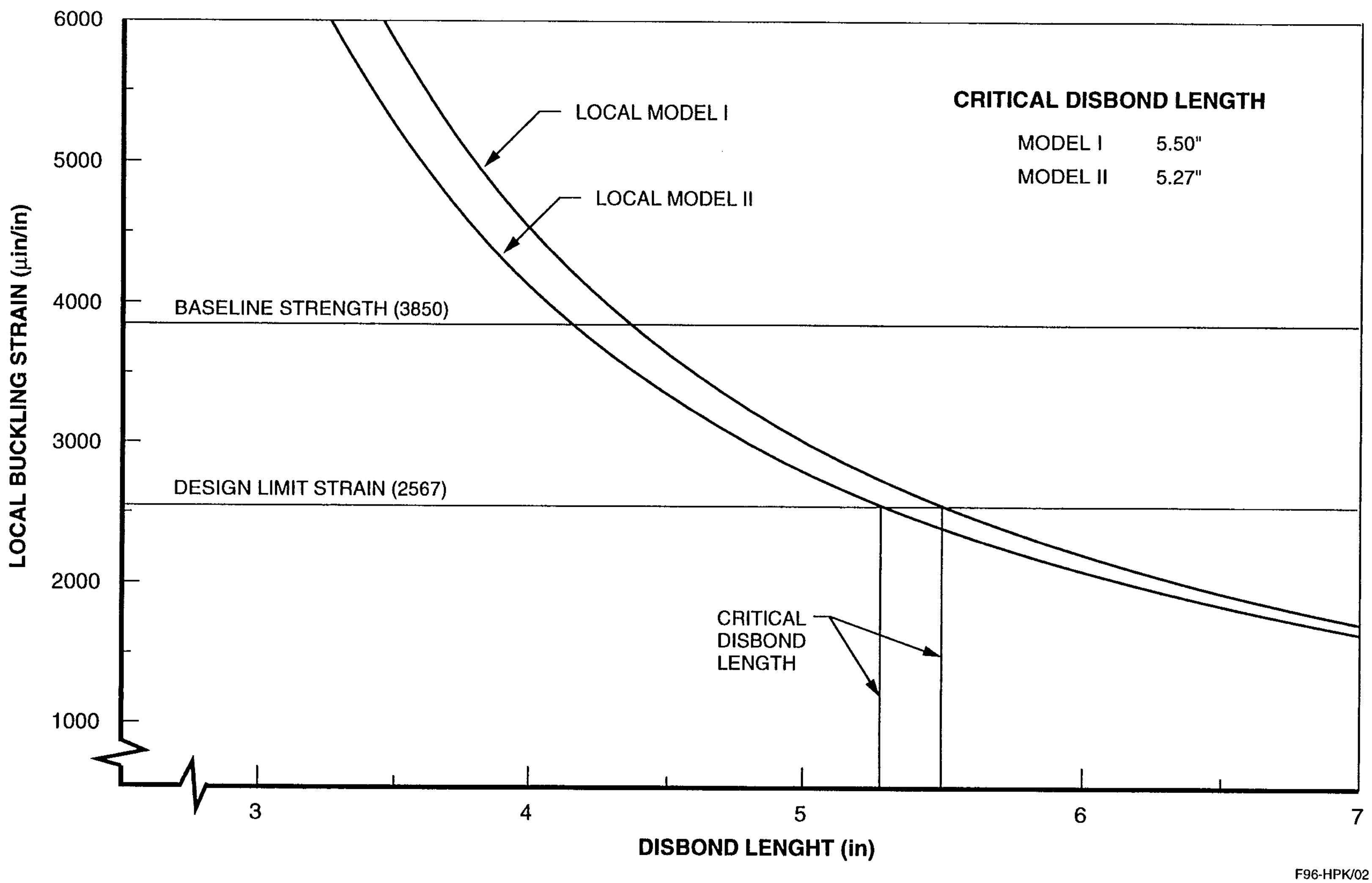


FIGURE 11. CRITICAL DISBOND LENGTH FOR THE INBOARD SECTION, SMALL-AIRPLANE WING.

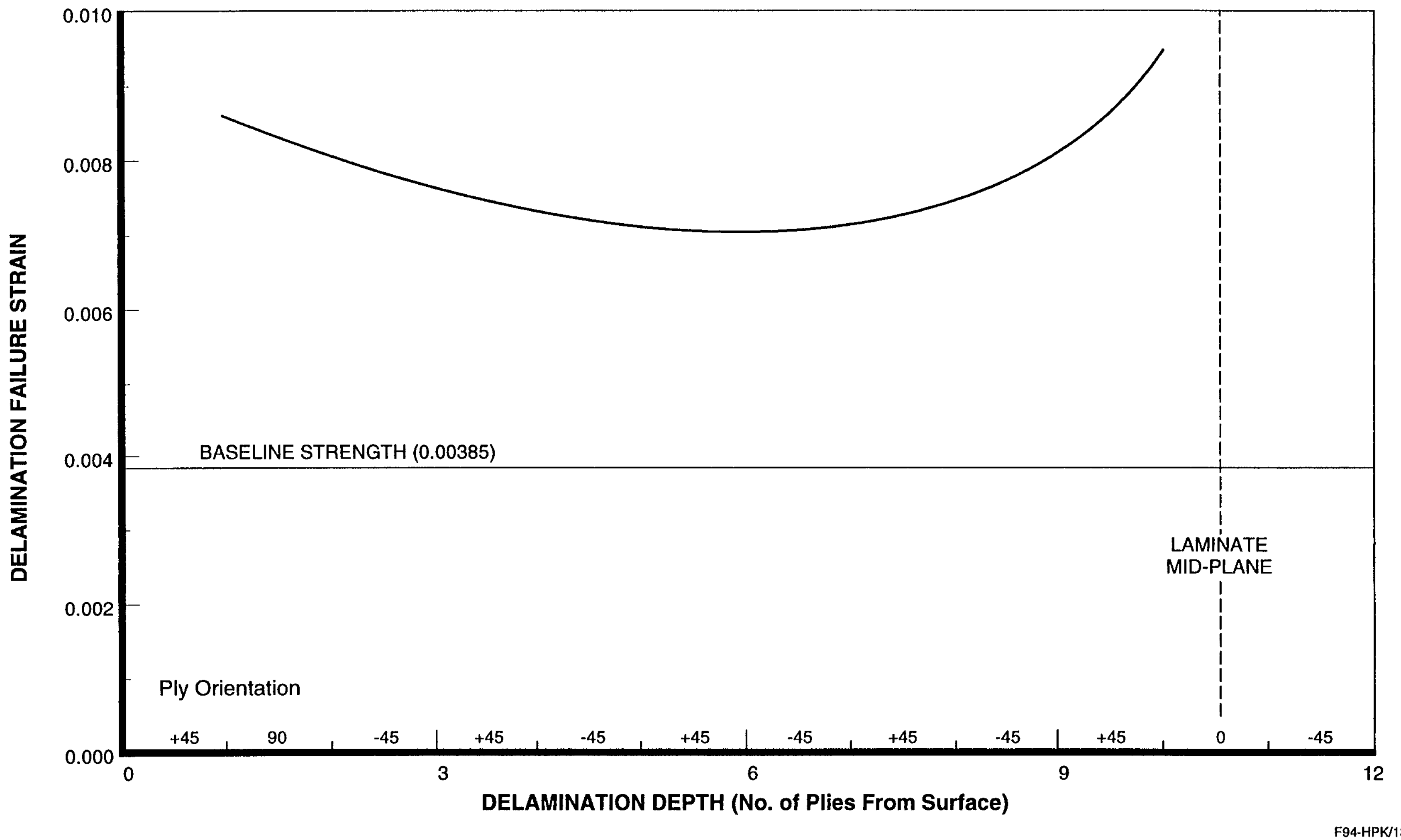


FIGURE 12. DELAMINATION STRENGTH AS A FUNCTION OF DELAMINATION DEPTH, TYPICAL SECTION, SMALL-AIRPLANE WING.

Similar results for the inboard section is shown in the following table.

TABLE 12. CRITICAL DELAMINATION SIZES FOR THE SMALL-AIRPLANE WING SKIN--INBOARD SECTION.

NO OF PLIES DELAMINATED	CRITICAL DELAMINATION DIAMETER (in)	FAILURE STRAIN	PANEL LOAD (kips)	STRAIN/ LOAD RATIO
1	5.54	0.00856	1,807	2.220
2	5.32	0.00785	1,658	2.038
3	4.96	0.00755	1,595	1.962
4	4.61	0.00731	1,544	1.898
5	4.25	0.00715	1,510	1.857
6	3.90	0.00714	1,509	1.855
7	3.55	0.00739	1,561	1.919
8	3.21	0.00807	1,704	2.095
9	2.87	0.00948	2,002	2.461

The residual strength as a function of the delamination depth is shown in figure 13. These results again indicate that delamination is a less severe damage scenario, and that impact damage tolerance design criteria are sufficient to account for this type of damage.

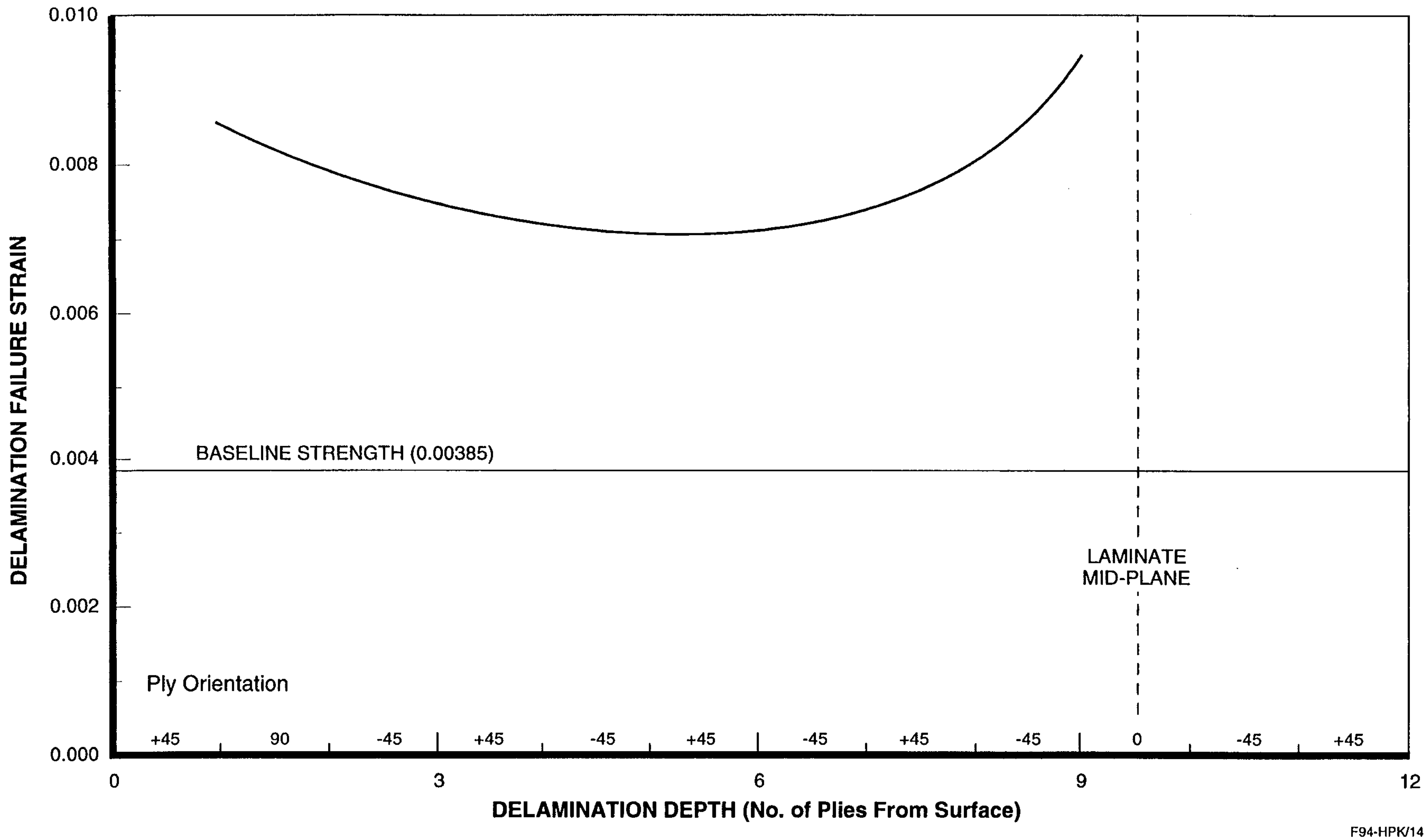


FIGURE 13. DELAMINATION STRENGTH AS A FUNCTION OF DELAMINATION DEPTH, INBOARD SECTION, SMALL-AIRPLANE WING.

SECTION 5

CERTIFICATION METHODOLOGY

The results of the present study were used to formulate an approach to certify composite aircraft structures with disbond type of damage. This approach was then integrated into the overall composite aircraft structural certification methodology developed in references 11, 12 and 18. The overall certification methodology is summarized in figure 14. The overall certification procedures for composite structures include three elements. These are (1) static strength, (2) durability, and (3) damage tolerance. The static strength and durability certification procedures are discussed in detail in reference 18, and the impact damage tolerance certification method is presented in reference 11. The procedures for assembly induced damage tolerance are detailed in reference 12. In the following paragraphs, the overall certification methodology is summarized and the procedures to certify structures with large area disbonds are discussed in detail.

5.1 STATIC STRENGTH CERTIFICATION

A building-block approach is adopted in reference 18 for both static and durability certification of composite structures. The testing requirements in this approach include design allowable, design development and full-scale testing. The details of the building-block approach are given in reference 19.

The purpose of the design allowable tests is to evaluate the material scatter and to establish strength parameters for structure design. Because composites are environmentally sensitive, design allowables should be obtained for the range of the environmental service conditions for an aircraft. Statistical analysis methods are used to compute the design allowables. MIL-HDBK-17B, reference 20, contains adequate guidelines for planning of design allowable testing, and these guidelines should be closely followed.

The philosophy for design development testing should be that the test environment used is the one that produces the failure mode which gives the lowest static strength. That is, the worst case environment, or the temperature associated with the most critical load should be used. The extent of the static test effort will be different from aircraft to aircraft and also from component to component. The levels of complexity in the design development testing should be functions of the design feature being validated and the predicted failure modes. Special attention should be given to correct failure mode simulation since failure modes are frequently dependent on the test

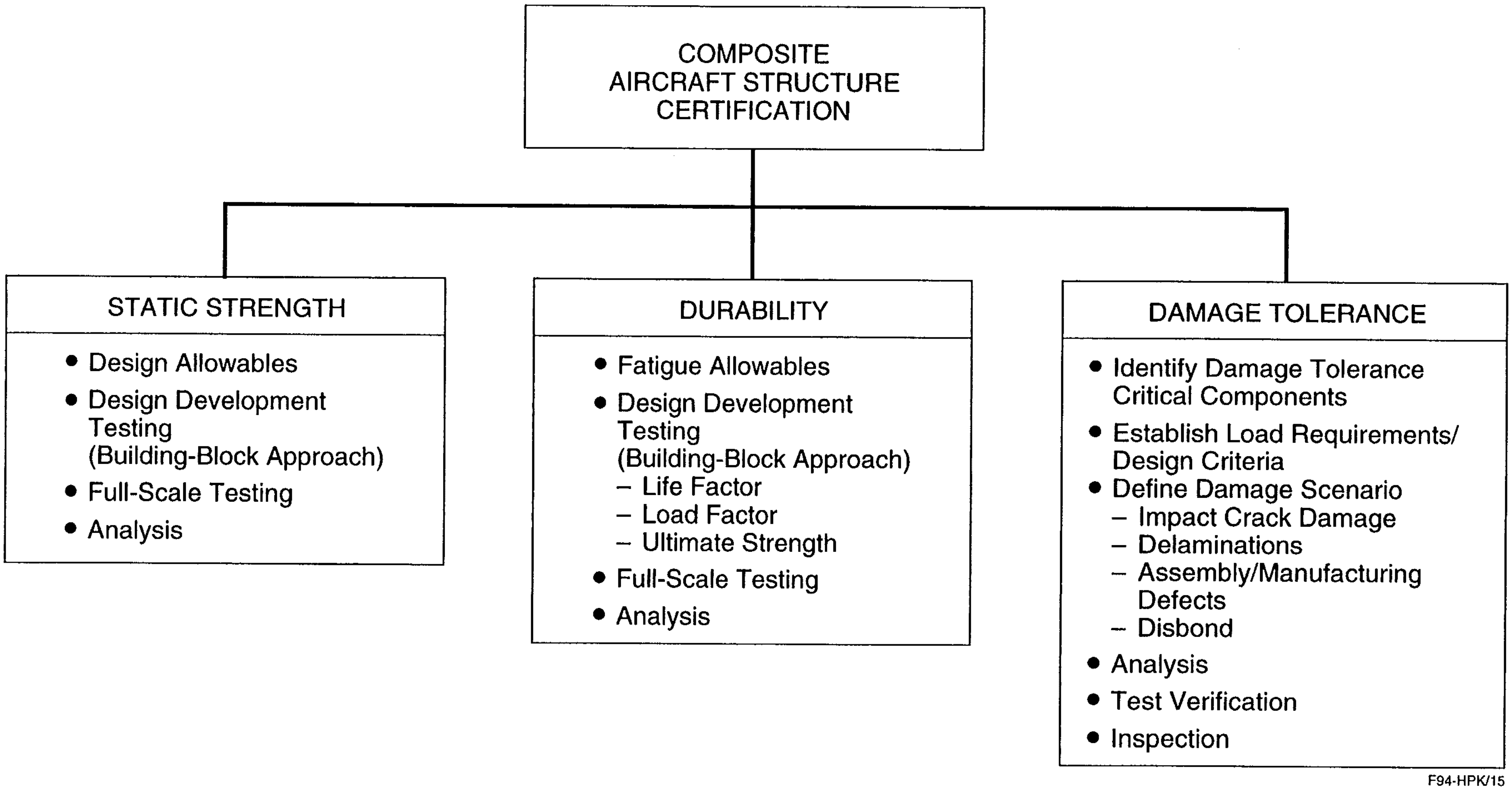


FIGURE 14. KEY ELEMENTS OF THE OVERALL COMPOSITE STRUCTURAL CERTIFICATION METHODOLOGY.

environment. In particular, the influence of complex loading on the local stress at a given design feature must be evaluated. In composites, out-of-plane stresses can be detrimental to structural integrity and, therefore, require careful evaluation. The number of replicates for each test should be sufficient to identify the critical failure mode and provide a reasonable estimate of the mean strength of the element and should be increased for the critical design features. If mixed failure modes are observed in a certain specimen type, more tests are required to establish the worst failure mode and the associated mean strength. A cost trade-off is usually involved in deciding the levels of complexity and the number of replicates.

The full-scale static test is the most crucial qualification test for composite structures for the following reasons. Secondary loads are virtually impossible to eliminate from complex built-up structure. Such loads can be produced by eccentricities, stiffness changes, discontinuities, pressure loading, and loading in the post-buckling range. Some of these sources of secondary loads are represented for the first time in the full-scale structural test article. These loads are not a significant design driver in metallic structures. However, the poor interlaminar strength of composites makes them extremely susceptible to out-of-plane secondary loads. It is very important, therefore, to carefully account for these loads in the design of composite structures.

In addition, a detailed correlation in terms of measured load, strain, structural analysis, and environmental effects between the design development and full-scale test data will be necessary to provide assurance of composite static strength. Static test environmental degradation must be accounted for separately either by adverse condition testing, by additional test design factors, or by correlation with environmental design development test data.

5.2 DURABILITY CERTIFICATION

The building-block approach is also recommended for durability certification of composite structures. The fatigue design allowables may be determined by the load factor approach, life factor approach, or the ultimate strength approach. Details of these approaches are contained in reference 18. In planning the fatigue allowable tests, the main consideration is the test environment. The test environment depends on the relationship between the load/temperature spectrum and the material operation limit. The recommended approach is to use simple, conservative constant temperature tests with a constant moisture level. The stress levels used in the fatigue tests should be selected so that the fatigue threshold can be established.

The environmental complexity necessary for fatigue design development testing will depend on the aircraft hygrothermal history. Three factors must be considered. These are structural temperature for each mission profile, the load/temperature relationships for the aircraft, and the moisture content as a function of the aircraft usage and structure thickness. In order to obtain these data, it is necessary to derive the real-time load-temperature profiles for each mission in the aircraft's history. These relationships will have a significant influence on the environmental fatigue test requirements.

As discussed in reference 18, the use of fatigue test data to verify fatigue life on subcomponents require long test duration because of the high fatigue life scatter observed in composite structures. The load enhancement factor approach or the ultimate strength approach is recommended in planning the fatigue design development testing. The number of replicates to be used in the fatigue design development testing should be determined using the same philosophy as in the static tests. A sufficient number of replicates should be used to verify the critical failure modes and to reasonably estimate the required fatigue reliability.

The work in reference 19 and other government sponsored programs have shown that composites possess excellent durability. In particular, the extensive database developed in Reference 19 showed that composite structures which demonstrated adequate static strength were fatigue insensitive. Therefore, reference 19 recommends that no durability full-scale test is required for all composite structures or mixed composite/metal structures with no fatigue critical metal parts, provided that the design development testing and full-scale static tests are successful. For mixed structures with fatigue critical metal parts, a two-lifetime ambient durability full-scale test will be required to demonstrate durability of the metal parts.

5.3 DAMAGE TOLERANCE CERTIFICATION

The key elements in damage tolerance certification of composite structures are shown in figure 14. The first element is to identify critical structural components. Guidelines for selection of structural components to be evaluated for damage tolerance capability are contained in AC 25.571-1A (reference 2) based mainly on experience with metal structures. In composite structures, the sensitivity to damage threat depends on the primary function of the component and the damage type. For example, impact damage or skin/stiffener disbond is a severe threat to compressively loaded components but is not as sensitive to tensile components. On the other hand, crack-like damage significantly degrades the strength of a tensile structural member but is not the most severe damage scenario for the compression members. Therefore, in addition to the conventional structural classification, critical damage tolerance components for composites should also be identified in terms of their primary loading modes. Damage scenarios and load requirements can then be defined according to the structural classification.

Damage tolerance of composite structures under compression loading has been extensively investigated (references 9, 11 and 12) and certification methodology has been adequately developed for this type of structure. References 5 and 6 investigated tension fracture of composite structures, but guidelines for certification have not been established. Guidelines and requirements for certification of composite structures subjected to combined loading and pressure loading are also not available at the present time. Development of damage scenarios and certification requirements for these types of structures would be important subjects of future research. In the following paragraphs, the damage tolerance certification approaches for impact damage, delaminations, manufacturing and assembly induced damage, and large skin/stringer disbond are summarized.

5.3.1 Impact Damage

The key elements of impact damage tolerance certification of composite structures are:

a. Testing requirements,

b. Impact threat definition,

c. Damage tolerance requirements,

d. Impact damage analysis.

The purpose of impact damage tolerance tests is to establish residual strength capability and strength scatter for damage tolerance analysis. Two levels of tests should be conducted on representative laminate coupons and structural elements. In planning the coupon tests a range of impact energy should be first identified. The range of impact energy depends on the laminate thickness and the material system. For composite materials commonly used in primary aircraft structures, the range of 20 to 100-ft-lb is appropriate. The impact damaged specimens are then tested for post-impact strength in compression. The number of specimens required for the coupon tests should be sufficiently large so that the trend of strength degradation and the scatter in strength can be confidently established.

Representative structural elements should be impacted and tested for residual strength in compression. The purpose of these tests is to determine the structural configuration effects on the residual strength.

A conservative impact threat should be used in the impact damage analysis. The impact threat distribution should be represented by a statistical function. In this functional representation, a modal impact energy and an energy level associated with a rare impact event are required. The medium impact threat proposed in reference 11 seems appropriate.

Impact damage tolerance design requirements are generally defined by mutual agreement between aircraft manufacturer and the certification agency. However, over conservatism of the requirements may result in a weight penalty. A sensitivity study conducted in Reference 11 has shown the effects of impact damage design requirements on the structural design. Further study in this area is needed.

The analysis methodology developed in reference 11 is an integrated methodology for damage tolerance evaluation of composite structures. This analysis method is recommended for certification for impact damage tolerance.

5.3.2 Manufacturing/Assembly Induced Damage

The capability of a structure to tolerate manufacturing or assembly induced damage must be addressed during the certification process to ensure adequate structural reliability. This is because final assemblies are not generally subjected to non destructive inspection, and even if an inspection is performed, not all areas are accessible after assembly. Based on an extensive survey of existing composite aircraft structures conducted in reference 21, the most degrading type of damage induced by a structural assembly is fastener hole damage. This type of damage affects structures with mechanically fastened elements. The strength of both tension and compression structural members are affected by this type of damage.

The results from the survey of reference 21 indicate that more than 90 percent of the damage is smaller than 2.0 inches in diameter. Therefore, it is recommended in reference 12 that 2-inch-diameter (or equivalent area) assembly induced damage should be assumed to exist in damage tolerance evaluation of mechanically fastened composite structures. Testing and analysis methods for structural evaluation recommended in reference 12 should be used to assure the integrity of the structure. Simple guidelines listed in reference 12 should also be followed to reduce the occurrence of assembly induced damage.

5.3.3 Interply Delaminations

Delaminations are a less severe damage type in terms of strength and life degradation, as compared to impact damage and assembly induced damage. The results of this study as well as the results of references 9 and 12, indicate that structures designed to comply with damage tolerance requirements for impact damage and assembly induced damage would adequately account for delaminations. Therefore, no additional requirements are recommended.

5.3.4 Skin/Stiffener Disbonds.

The results of the current study indicate that, for typically designed wing structures, a completely disbonded stringer represents the most severe damage scenario among the damage types considered. This type of damage mainly affects bonded or cocured structures under predominantly compression loads. The local strength at the damaged location depends on the design details of the structure. The local failure for structure with a completely disbonded stringer can be reduced to as low as 17 percent of the residual strength due to impact damage. Because of the large strength reduction, damage tolerance design based on such a damage scenario would impose a significant weight penalty to the structure. In order to achieve an efficient structure design without sacrifice the structural integrity a partial skin/stringer disbond is recommended as a damage tolerance certification requirement. The following damage tolerance certification procedures are recommended.

a. Design the structure to comply with the impact damage tolerance requirements. The impact damage tolerance certification procedures outlined previously should be used as a baseline.

b. Analytically establish the maximum disbond length. The damaged structure with this disbond length should be able to withstand limit load as the size of the delamination is large. Because compression loading is the most critical condition for structures with disbond type of damage, local stability should be the key consideration in the analysis. The local buckling models discussed in Section 4 are recommended in defining the maximum disbond length.

c. Perform nondestructive inspection to assure that no initial defects exceed the analytically established maximum disbond size.

d. Implement design features to limit the damage size. The rib spacing should be adjusted to limit the disbond to within the established maximum length, if no significant weight impact results. Properly spaced fasteners may be used to assure that the local structural response is confined.

e. Conduct verification tests to assure that no undesired local deformation occurs due to large skin/stringer disbond.

SECTION 6

CONCLUSIONS AND RECOMMENDATIONS

6.1 SUMMARY

The results of this research program are summarized below:

a. Two existing composite aircraft wing structures, one representative of a large airplane and the other a small airplane, have been selected for damage tolerance evaluations.

b. Three damage scenarios have been considered in the damage tolerance evaluations. They are impact damage, delaminations, and skin/stiffener disbonds.

c. Residual strengths based on impact damage design of the structures have been used as the baseline strength capability of the structures.

d. Residual strength ratios, in relation to the baseline strength, have been analytically determined for structures with delaminations, partial disbond, and complete disbond.

e. Structural certification procedures for the delaminations and disbond damage scenarios are recommended.

f. Certification procedures has been integrated into a complete certification methodology.

6.2 CONCLUSIONS

The following conclusions may be drawn from the investigations undertaken in this program.

a. Large area interply skin delamination is a less severe damage threat to composite structures. Structural design based on impact damage requirements properly accounted for this damage type; no additional requirements are needed.

b. Local, instability related failures are the dominant failure modes for bonded or cocured structures with skin/stiffener disbond type of damage.

c. The residual strength of structures with partial or complete skin/stiffener disbond depends on the design details of the structure.

d. A complete disbond of a stringer from a structure represents the most severe damage scenario among the damage types considered.

e. A damage tolerance requirement based on the damage scenario that a stiffener completely disbonded would result in unacceptable weight and structural efficiency.

f. A partial disbond can be used as a damage tolerance design requirement. The maximum length of the disbond should be determined so that the local strength is comparable to the residual strength based on impact damage. This maximum disbond length can also be used to establish detail design requirements for damage containment.

g. Inspections, test verification and additional design features may be required to assure the structural integrity.

6.3 RECOMMENDATIONS

Substantial progress has achieved through the work of references 11, 12, 18 and the present investigation in developing a certification methodology for composite structures. The following work for further development and validation of the methodology are recommended.

a. Develop general guidelines for selection of damage tolerance design criteria. A total of seven different impact damage tolerance design criteria were used in a sensitivity study in Reference 11. The results indicated that design criteria significantly influence the structural design. These type of sensitivity studies should be conducted to further examine the impact of design criteria on structural weight and cost.

b. Fully develop damage scenarios for damage tolerance certification of composite structures. The damage scenarios should be developed based on service experience. Critical damage scenarios should be established in accordance with structural type, loading conditions, and environments.

c. Develop general guidelines for damage tolerance evaluation of structures subjected to tension loading and combined mechanical and pressure loadings.

d. Fully integrate the strength/stiffness, durability, and damage tolerance certification methods so that risk assessment and trade studies can be performed in structures/materials selection for certain design concepts.

e. Investigate the validity of the current certification methodology on structures using new composite materials and new fabrication processes.

f. Evaluate the weight and cost impact of the damage tolerance requirements on future aircraft programs using composite materials.

REFERENCES

1. Federal Aviation Regulations, Part 25, Airworthiness Standards: Transport Category Airplanes, Department of Transportation, Federal Aviation Administration, 1-1-92 Edition.

2. Advisory Circular AC 25.571-1A, Damage-Tolerance and Fatigue Evaluation of Structure, Federal Aviation Administration, March 1986.

3. Ilcewicz, L. B., Smith, P. J., and Horton, R. E., "Advanced Composite Fuselage Technology," Third NASA Advanced Composites Technology Conference, NASA CP-3178, pp. 97-156, 1992.

4. Hawley, A. V. and Sutton, J. O., "Design and Analysis Considerations for Stitched/RTM Composite Wing Structures," Fourth NASA/DoD Advanced Composites Technology Conference, NASA CP-3229, pp. 421-452, 1993.

5. Walker, T. H., Ilcewicz, L. B., Polland, D. R., and Poe, CC., Jr., "Tension Fracture of Laminates for Transport Fuselage - Part 2: Large Notches," Third NASA Advanced Composites Technology Conference, NASA CP-3178, pp. 727-758, 1992.

6. Walker, T. H., Ilcewicz, L. B., Polland, D. R., Bodine, J. B., and Poe, C. C., Jr., "Tension Fracture of Laminates for Transport Fuselage - Part 3: Structure Configurations," Fourth NASA/DoD Advanced Composites Technology Conference, NASA CP-3229, pp. 243-263, 1993.

7. Dost, E. F., Avery, W. B., Ilcewicz, L. B., and Grande, D. H., "Impact Damage Resistance of Composite Fuselage Structure, Part I," Ninth DoD/NASA/FAA Conference on Fibrous Composites in Structural Design, DOT/FAA/CT-92-25, Volume II, September 1992.

8. Dost, E. F., Avery, W. B., Finn, S. R., Ilcewicz, L. B., and Scholz, D. B., "Impact Damage Resistance of Composite Fuselage Structure, Part 3," Fourth NASA/DoD Advanced Composites Technology Conference, NASA CP-3229, pp. 205-229, 1993.

9. Horton, R. E., Whitehead, R. S., et al., "Damage Tolerance of Composites," Volumes I, II and III, Report No. AFWAL-TR-87-3030, July 1988.

10. Soderquist, J. R., "Design/Certification Considerations in Civil Composite Aircraft Structure," SP-727, Aerospace Technology Conference and Exposition, SAE, October 1987.

11. Kan, H. P., Cordero, R., and Whitehead, R. S., "Advanced Certification Methodology for Composite Structures," Final Report, Contract No. N62269-87-C-0259, Naval Air Development Center, September 1989.

12. Kan, H. P., "Delamination Methodology for Composite Structures," Final Report, Contract No. N62269-90-C-0282, Naval Air Warfare Center, February 1993.

13. Rosenzweig, E., "U. S. Navy Survey of Damage and Defects in Advanced Composite Components in Service," NADC, 1985.

14. Shyprykevich, P., "Empennage Structure Service Difficulties, Year 89-93," Private Correspondence, FAA Technical Center, 1993.

15. Advisory Circular AC 20-107A, Composite Aircraft Structures, Federal Aviation Administration, April 1984.

16. Lincoln, J. W., "Certification of Composites For Aircraft," Proceedings of the USAF Aircraft Structural Integrity Program Conference, Sacramento, CA, December 1986.

17. Lekhnitskii, S. G., Anisotropic Plates, 2nd Edition, Translated from Russian by S. W. Tsai and T. Cheron, Gordon and Breach, 1968.

18. Whitehead, R. S., Kan, H. P., Cordero, R., and Saether, E. E., "Certification Testing Methodology for Composite Structures," Report No. NADC-87042-60, October 1986.

19. Whitehead, R. S., et al., "Composite Wing/Fuselage Program," AFWAL-TR-88-3098, Volumes I-IV, February 1989.

20. MIL-HDBK-17B, Plastics for Aerospace Vehicles Part 1. Composite Materials for Aircraft and Aerospace Applications, Volume I Guidelines.

21. Brachia, C., Wenthal, S., and Goering, T., "Survey of Assembly Induced Delaminations in Composite Structures and Related Analysis," Report No. NAWCADWAR-92057-60, July 1992.